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The Development of Cryogenic Storage Systems for Space Flight

NATIONAL AERONAUTICS AND SPACE ADMINISTRATION U.S.A.

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Bay 4 of the Apollo 13 service module with the cryogenic storage systems installed. The two skirted spherical hydrogen tanks are on the lower shelf and the two spherical oxygen tanks are on the upper shelf.

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The Development of Cryogenic Storage Systems for Space Flight

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Foreword

The staff of the Power Generation Branch of the Propulsion and Power Division of the NASA Manned Spacecraft Center has managed the development of cryogenic gas storage systems for space flights since the establishment and staffing of the Manned Spacecraft Center in Houston, Texas. Traditionally the management of this technology development has been done in an atmosphere of cost and technical competition. As a result, several companies currently are qualified to supply cryogenic gas storage systems or have provided these systems for the U.S. space efforts. The guidelines that have been used are common to the entire space program, and they include an orderly developmental process. A building-block approach has been used that involves a process of system-requirements determination, research to obtain technology advances to meet the system requirements, reduction of new concepts to practice, and development of the necessary manufacturing capability. This approach has been and is being used to produce developmental and flight hardware. Throughout this process, there has been a requirement to accomplish these tasks in the most efficient way possible at minimum cost and time, consistent with producing a system that meets all of the necessary requirements.

The progress in the development of cryogenic gas storage systems during the last few years has been documented by the personnel of the Power Generation Branch and the contractor personnel that have been involved in individual contracts. Primarily these reports have been technical in nature and have been limited to a discussion of a specific system or program. Accordingly this report is intended to serve the following purposes in a comprehensive manner from the historical and informative point of view: (1) to trace and report on the development of cryogenic technology from its inception through the Gemini Program, Apollo Program, Apollo Applications Program (Skylab), and beyond to some estimates of the role of cryogenics in the future; (2) to provide the information necessary to understand the interdependence and relationship of the various cryogenic systems that have been developed and the programs that have been involved; (3) to provide fundamental information necessary for a basic understanding of cryogenics; and (4) to present a partial compilation of the uses that are being made of aerospace cryogenics technology spinoff.

Contents

1 Basic Concepts and History of Cryogenics

The vapor that billows gracefully from the umbilical connections of a Saturn launch vehicle on a Cape Kennedy launch pad has become a familiar sight to millions of television viewers. The source of that vapor is a frigid fluid that makes the launch possible. When combined with RP–1, a hydrocarbon fuel, the chemical energy contained in the cryogenic oxygen is the source of the 7.5 million pounds of thrust that is required to propel the Saturn vehicle from the launch pad into space. Cryogenic oxygen and hydrogen in the S–IVB stage are the oxidizer and the fuel, respectively, that are required to produce the 225 000 pounds of thrust that is available from the third stage of the Saturn launch vehicle. The three crewmen in the command module depend on the cryogenic storage of oxygen for breathable atmosphere, and they depend on the cryogenic storage of oxygen and hydrogen for the primary power source (fuel cells) during the mission.

As a direct result of international efforts in the exploration of outer space, the field of cryogenics has undergone a phenomenal growth during the past decade. Cryogenics was one of many fields in which major scientific breakthroughs were required in support of the space effort. Now, cryogenic fluids have become useful servants of man in many fields of endeavor ranging from the space program to medicine. In this report, the progress of cryogenics is traced from its simple beginning to the complex field that it is today, and primary consideration is given to spacecraft cryogenics because of the experience of the authors in this area of cryogenics. Numerous efforts and achievements were necessary, however, to produce a body of information that could be developed by aerospace researchers to its present state. As a comprehensive report, a developmental history of cryogenics that did not include a basic acknowledgement of the early efforts would be remiss. The purpose of chapter 1 is to meet this basic requirement and to provide insight into background information for the material to be covered in the other chapters.

NOMENCLATURE

Several words and terms used in this report may be unfamiliar, either in meaning or in context, to some readers. The following definitions and explanations of terms should prevent misunderstanding.

The word *cryogenic* means "suitable for production by icy cold conditions." It comes from the Greek word *kryos*, which means icy cold. Cryogenics is the discipline that involves the properties and use of materials at extremely low temperatures; it includes the production, storage, and use of cryogenic fluids. A gas is considered to be a cryogen if it can be changed to a liquid by the removal of heat and by subsequent temperature reduction to a very low value. The temperature range that is of interest in cryogenics is not defined precisely; however, most researchers consider a gas to be cryogenic if it can be liquefied at or below −240° F. The most common cryogenic fluids are air, argon, helium, hydrogen, methane, neon, nitrogen, and oxygen. Other gases that are being used in space probes and high-energy liquid-propellant rockets include fluorine and nitrogen trifluoride.

Gases that can be liquefied at temperatures greater than −240° F and that remain liquid under pressure at ambient conditions without refrigeration are considered to be noncryogenic. Chlorine and the Freon refrigerants are examples of noncryogenic gases. Normally, cryogenic fluids exist as gases at ambient temperature and pressure. Therefore, refrigeration and insulated or supercooled conditions are required for these fluids to remain liquid for any appreciable period of time. Generally this is accomplished by storing the cryogen in a vacuum-insulated storage vessel called a dewar, or in a double-jacketed tank that contains a surrounding layer of another cryogenic fluid (for example, liquid nitrogen, LN_2). Cryogenic fluids are compressible, whereas noncryogenic fluids are not compressible. Oxygen compressibility characteristics are shown in table 1, and

TABLE 1—*The Required Storage Volume for 2 Pounds of Oxygen under Various Storage Conditions* [a]

Storage pressure, psia	Storage temperature, ° F	Required storage volume, ft³
14.7	70	24.16
900	70	.38
900	−297	.03

[a] The metabolic-oxygen requirement for one man is approximately 2 lb/24 hr.

some indication is given of the desirability of the cryogenic storage of gases when volume is a major design constraint.

It will be noted that the phrase, "cryogenic gas storage system" (CGSS), is used almost exclusively throughout this document. The reasons for this are as follows.

(1) The more common term "cryogenic storage system" is inexact in that the system itself is not cryogenic (only fluids can be cryogenic).

(2) Supercritical fluid-storage systems begin each delivery cycle containing compressed liquid rather than supercritical fluid by the strict definition. However, because the fluid behaves as a high-density gas in either the supercritical or compressed-liquid region, the word *gas* is descriptive.

(3) The function of all of the flight systems described in this document is to supply the user system with the usual low-density gases.

The point of a phase diagram of a pure substance that corresponds to its critical state is referred to as the critical point of that substance. The temperature, pressure, and specific volume at the critical point are called the critical temperature, critical pressure, and critical volume, respectively. The critical state is a state in which the liquid and vapor phases of the substance have the same density. The triple point of a substance defines the state in which all three phases (solid, liquid, and vapor) may be present in equilibrium.

TEMPERATURE SCALES

Temperature scales have been determined by the use of discrete temperature values which occur in nature and which can be repeated under laboratory conditions. Two of the four temperature scales that are in common use were determined in that manner. The Celsius scale (hereafter called the centigrade scale) is based on the points at which water, at standard pressure, may be frozen (0° C) and boiled (100° C). On the Fahrenheit scale, 32° F is the freezing point of water and 212° F is the boiling point of water.

Based on the second law of thermodynamics, a temperature scale can be defined that is independent of any thermometric substance; usually, this absolute scale is referred to as the thermodynamic scale of temperature. Two other temperature scales, Kelvin and Rankine, are absolute scales and are used most often in the measurement of low temperatures. This is because of the requirement for absolute-temperature values in some calculations, and perhaps because of a preference for positive rather than negative numbers. The Kelvin scale is an extension of the centigrade scale. On the Kelvin scale, the primary point of reference is absolute zero (0° K), which is equivalent to —273.16° C. The Rankine scale is an extension of the Fahrenheit scale. The Rankine scale starts at absolute zero (0° R), which is equivalent to —459.69° F. The common temperature scales that are in use today are shown in figure 1.

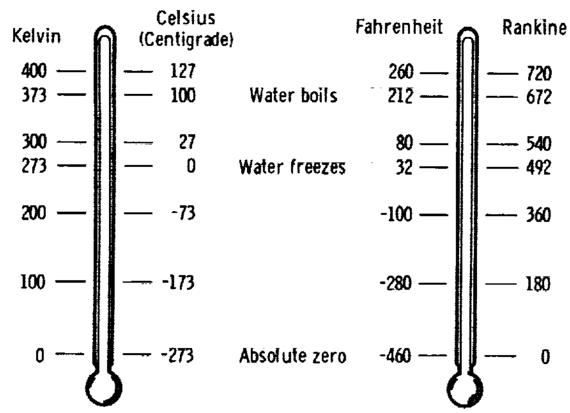


FIGURE 1.—Temperature scales that are in common use.

In the preceding discussion, the term "absolute zero" has been mentioned several times. The existence of absolute zero has been known since it was determined that a 1/273 decrease in the pressure of a gas would result from each centigrade-degree temperature decrease. A graphic-extension method for determining absolute zero is shown in figure 2. If the decreasing-pressure line is extended so that it reaches the zero-pressure point, it does so at a point that corresponds to —273° C. The pressure cannot decrease beyond this point; therefore, the temperature cannot de-

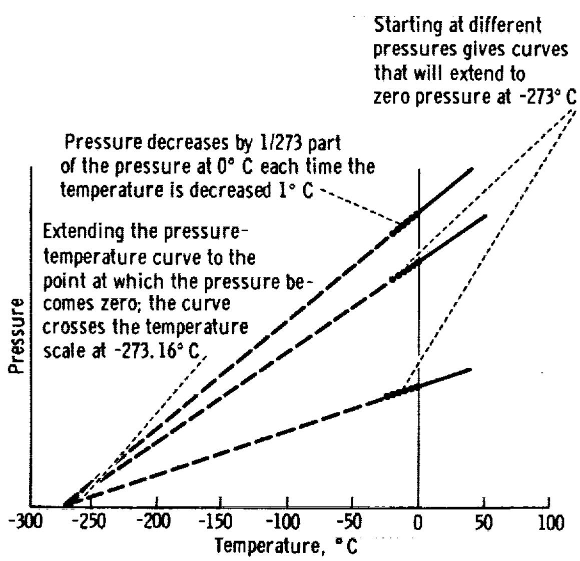


FIGURE 2.—Graphical method for the determination of absolute zero.

crease farther. Hence, —273° C must correspond to an absolute zero. If the experiment is repeated with a different gas or is started at a different pressure, the extended line still crosses at —273° C. Therefore, absolute zero is the lowest temperature that is predictable by means of theory. It cannot be said that it is the lowest temperature that could be attained, because the theory also involves the concept that absolute zero cannot be attained, although it can be approached extremely closely.

THE PROCESS OF LIQUEFACTION

Most of the early years of cryogenics were concerned with the liquefaction of gases. Some attention was directed to the preservation of the cryogens once they were obtained, but liquefaction was the first problem to be solved. The success of Onnes in liquefying helium was the final confutation of the concept of permanent gases. However, the initial methods of gas liquefaction were inefficient, costly, and productive of quantities that were suitable only for laboratory use. The next problem to be solved was that of improving the existing gas-liquefaction methods so that cryogenics could be of practical use and not just be a laboratory curiosity. This problem has been solved. Today, a cryogenics industry exists that produces large quantities of cryogens at a reasonable cost. The existence of this industry is proof that many of the original problems and inefficiencies have been eliminated or minimized. A discussion of the basic processes of liquefaction should result in a better understanding of this aspect of cryogenics.

Gas Liquefaction

Gas liquefaction is achieved by cooling the gas to the condensation temperature and then removing the latent heat of vaporization. Therefore, the only basic requirement is a suitable method of refrigeration; that is, a refrigerative process that is effective in removing heat at a sufficiently low temperature. If it were possible to use an ideal refrigeration process that had no losses, the energy required for gas liquefaction would be considerably less than the energy required by the best existing liquefiers. The inability to achieve the ideal process is a result of the inefficiencies of the practical refrigeration processes and the imperfect methods that are used to conserve refrigeration efficiency.

In this section the ideal-gas-liquefaction process and the fundamental methods of gas-liquefaction are discussed. As a rule, the cryogenics researcher is not concerned with the details of liquefaction because he can purchase cryogens or a turn-key liquefaction unit; therefore, no attempt will be made to present detailed descriptions. Some knowledge of the concepts that are involved in liquefaction is helpful, though not required, for an understanding and appreciation of cryogenics.

Ideal-Gas-Liquefaction Process

An ideal-gas-liquefaction process will result in the liquefaction of a unit quantity of gas with the least possible expenditure of energy. Theoretically, the optimum method of achieving the ideal would be by the use of a process that is reversible thermodynamically. Such a process is indicated in the temperature-entropy diagram in figure 3. It is assumed that the process is of a continuous-flow nature. A gas enters a compressor at a certain initial pressure and is compressed isothermally; the heat caused by compression is absorbed by a heat sink. Isothermal compression is followed by an isentropic-expansion phase. The energy produced by the expansion engine is returned to help with subsequent isothermal compression. The pressures are chosen so that liquid will be produced after the expansion. This process bears little resemblance to that used in any liquefier thus far devised, because perfect isothermal compression and isentropic expansion are not attainable. Also, an extremely high pressure would be needed to compress the gas to such a density that, after isentropic expansion, the gas would be converted completely to liquid.

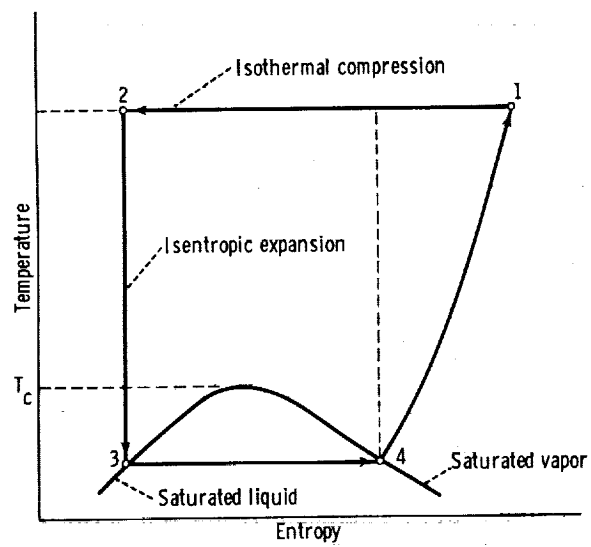


FIGURE 3.—Idealized refrigeration cycle.

Practical Liquefaction Methods

Fundamentally, three dissimilar methods exist for the production of the refrigeration that is necessary for gas liquefaction. These principal methods are the cascade process, Joule-Thomson expansion, and the expansion of a gas in an engine that is doing external work.

Cascade process.—The cascade process involves a range of substances in which the critical point of one substance is higher than the triple point of another substance. Typically substances that are used are fluids such as Ucon and Freon refrigerants, ammonia, or any other refrigerant that will condense at or near ambient temperature under moderate pressure. The cascade method will produce temperatures as low as −100° F, and it is used in most commercial refrigeration systems.

To liquefy a gas by means of compression, the temperature of the gas must be below the critical temperature—a requirement that was first recognized by Thomas Andrews in 1863. Thus, the cascade process first involves the compression of a gas to a pressure that is sufficiently high for the gas to condense at the temperature of an external cooling substance (heat sink). Then, the liquid that is produced is expanded to a lower pressure. During this expansion, part of the liquid vaporizes. This vaporization removes heat from the liquid and causes the remaining liquid to be cooled. This cooled liquid is then vaporized at constant temperature and pressure to provide refrigeration for the next stage of the liquefier.

The process just outlined is repeated for each subsequent stage of the liquefier. Generally the first stage of a cascade system uses water as the cooling substance. The incoming vapor of each subsequent stage is precooled by the refrigeration produced in the preceding stage. Each stage produces a liquid that is cooler than the liquid that was produced by the previous stage. A schematic of a three-stage cascade system is shown in figure 4. The cascade process is of industrial importance because it approaches the ideal thermodynamically reversible process more closely than does any other process. However, it does have one important limitation: It cannot be used to liquefy helium or hydrogen because of the lack of suitable substances to bridge the gaps in the required temperature range.

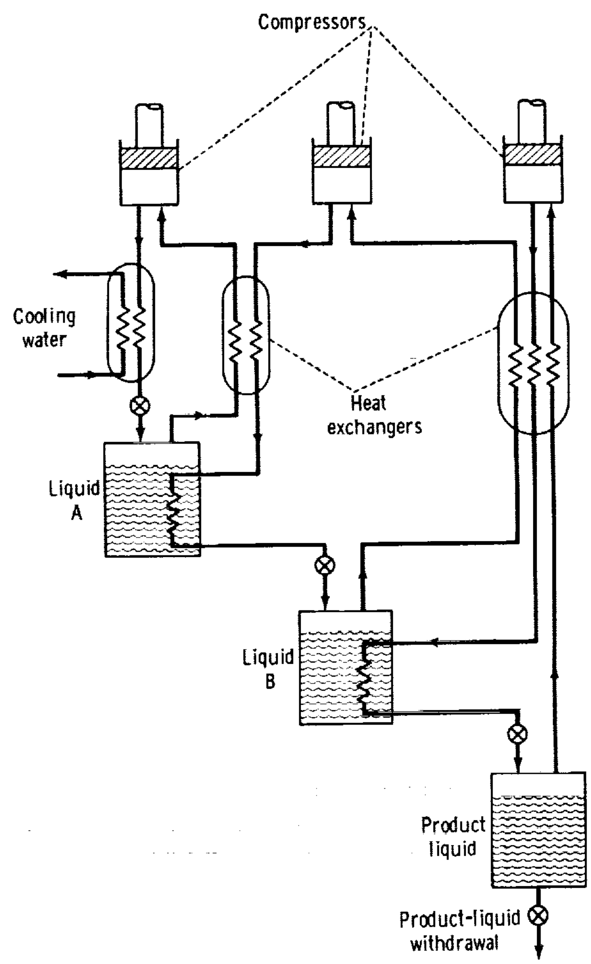


FIGURE 4.—Three-stage cascade system.

Joule-Thomson expansion. — The Joule-Thomson expansion method is an internal-work process that is dependent upon the Joule-Thomson effect to produce low temperatures. In this process, a gas is compressed to a high pressure; the heat of compression is removed; and then the gas is expanded through a valve, known as a Joule-Thomson (J–T) valve, to a low-pressure receiver. The expanding gas does not do any external work, such as moving a piston, but it does perform internal work in the form of expansion against molecular forces of attraction. Under the proper thermodynamic conditions, this energy expenditure results in a slight cooling of the gas and eventually results in liquefaction. After expansion, the cold gas is returned to the compressor through a countercurrent heat exchanger that cools the incoming high-pressure gas. This results in a cumulative cooling effect on the gas and eventually results in liquefaction. A simplified schematic of liquid-air production by means of Joule-Thomson expansion and the corresponding temperature-entropy diagram for the process are shown in figure 5.

Joule-Thomson expansion can be used to liquefy all gases. However, hydrogen, helium, and neon must be precooled sufficiently before they are expanded. To understand why it is necessary to precool these gases, one must investigate the proper thermodynamic conditions under which Joule-Thomson throttling results in cooling for a particular gas. If the gas is at a temperature that is greater than the inversion temperature when it is expanded, it will be warmed by the expansion process. If the gas temperature is less than the inversion temperature, a cooling effect will be produced by the expansion. Therefore, the expansion temperatures must be less than the inversion temperature to incur the cooling effect. A list of the inversion tempera-

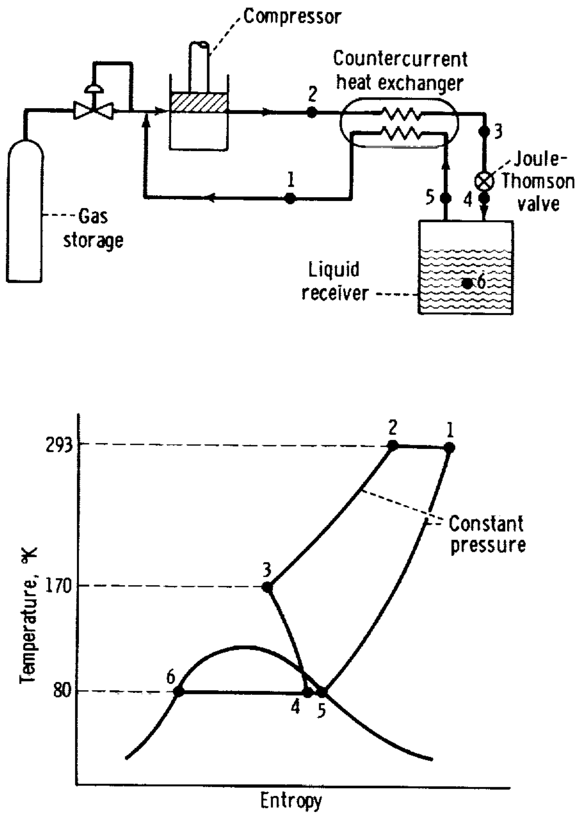


FIGURE 5.—Schematic of liquid-air production by the use of Joule-Thomson expansion and the corresponding temperature-entropy diagram.

tures of several common gases is given in table 2.

TABLE 2—*The Maximum Inversion Temperatures of Some Common Gases*

Gas	Inversion temperature, °R	Inversion temperature, °F
Carbon dioxide	2700	2240
Oxygen	1370	910
Argon	1300	840
Nitrogen	1120	660
Air	1085	625
Neon	450	−10
Hydrogen	364	−96
Helium	72	−388

It may be noted that hydrogen, helium, and neon are not below their inversion points at room temperatures. Thus, precooling these gases before expansion is necessary to obtain refrigeration from the Joule-Thomson process.

External-work method. — The external-work method of liquefaction is important for gases (for example, hydrogen) that have inversion temperatures that are considerably less than ambient temperature. In 1902, Claude used this method in conjunction with Joule-Thomson throttling for the liquefaction of air. The external-work method involves the use of a low-temperature expansion engine to increase the amount of refrigeration obtainable from a given pressure ratio. A gas is precooled slightly, compressed, and then the high-pressure gas is split into two streams. One of the streams is expanded in an engine doing external work, whereas the other stream receives additional cooling as it goes through a countercurrent heat exchanger and then is expanded through a Joule-Thomson valve into a low-pressure receiver. The gas stream that was expanded in the engine leaves the engine at reduced temperature and pressure. Then the gas is routed through the heat exchanger to provide cooling for the incoming high-pressure gas stream. The high-pressure gas stream is cooled sufficiently to lower the temperature and produce partial liquefaction when it is expanded through the Joule-Thomson valve. This process can be used to liquefy all gases, including hydrogen, helium, and neon. A representation of the process is shown in figure 6.

Other processes.—The methods of liquefaction just discussed are the bases for the liquefiers of the past and the present. The Linde and Hampson liquefaction processes involve the Joule-Thomson effect, whereas the Claude and Heylandt processes involve the external-work method in conjunction with Joule-Thomson throttling. Two other liquefaction processes that merit discussion are irreversible adiabatic expansion and adiabatic demagnetization.

A noteworthy adaptation of the processes just discussed is that of irreversible adiabatic expansion. This method was used by Cailletet to liquefy oxygen and by Simon to liquefy helium. In this process, advantage is taken of the work done on the high-pressure

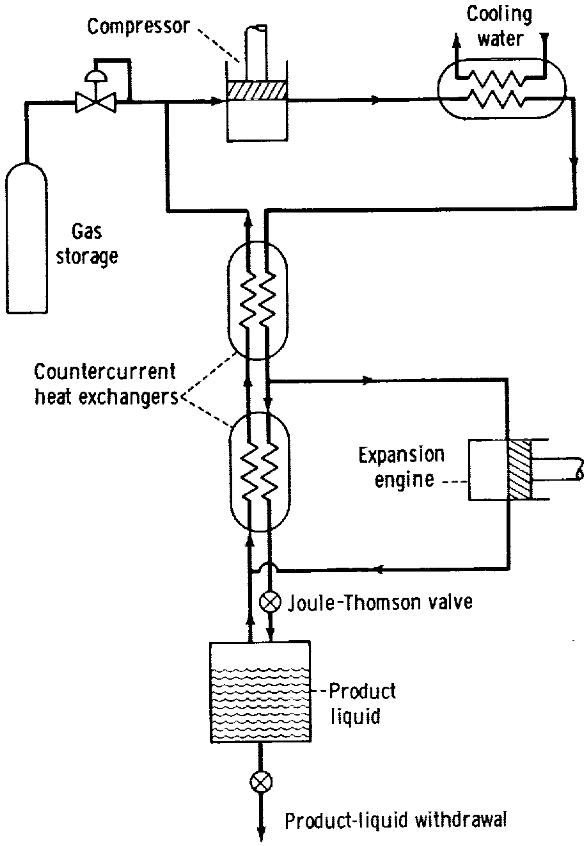


FIGURE 6.—The external-work method of liquefaction.

side of an expansion valve. After expansion, the gas that remains in the formerly high-pressure region has done work in the expulsion of the other gas molecules from the high-pressure region. If this same region is insulated sufficiently, this expenditure of energy will cool the gas. The cooling effect is enhanced by the deviations from ideal-gas behavior. In the Simon helium liquefier, helium gas is precooled with solid hydrogen to approximately 10° K at a pressure of 150 atmospheres. After thermal equilibrium is achieved, the gas is bled off through an expansion valve until the pressure in the chamber is reduced to 1 atmosphere. When the pressure has been released, the chamber is approximately 80 percent full of liquid.

A method for further reduction of the low temperatures achieved by the methods discussed previously is the adiabatic demagnetization of a paramagnetic salt. A paramagnetic substance will increase in heat when it is placed in a magnetic field, and it will cool when it is demagnetized. The heating and cooling effect is slight and is noticeable only at extremely low temperatures. If another method is used to cool a paramagnetic substance to 1° K and then a magnetic field is applied, the temperature of the substance will increase. Then, if a refrigeration system is used to return the temperature to 1° K, subsequent demagnetization will result in a temperature reduction. By a two-stage process, this method has been used to obtain temperatures as low as one millionth of a degree absolute. Demagnetization at the temperature of liquid helium (LHe) constitutes the first stage. A second sample of paramagnetic salt is cooled in a magnetic field by the refrigeration produced in the first stage and is then demagnetized adiabatically.

Because of many achievements in the production, storage, and transportation of cryogenic fluids, sufficient quantities of these fluids are available now to scientific and industrial institutions to meet their demands. Obtaining these fluids is as easy as having fresh milk delivered to the home. In the case of liquid air and nitrogen, even the cost per gallon is comparable.

HEAT TRANSFER: THE PROCESS TO BE CONTROLLED

The end of the first century of cryogenics is near. During the first half of that century, primary emphasis was placed on the production of cryogens. During the last half of that century, however, emphasis has been placed on the preservation and use of cryogens. Thus the transfer of heat from the surrounding environment to a contained cryogen has become a key consideration and is one of the main concerns of the modern cryogenics engineer. In most practical cryogenic applications, changes in the environment are prohibitive; therefore, the environment exists as an infinite heat source. Thus, the container has been the main consideration in the effort to limit the transfer of heat.

Protecting a mass of liquid nitrogen from an ambient environment is comparable to keeping an ice cube from melting in a hot oven. To accomplish either of these goals, a special container must be used. The design and development of this special container are the major subjects to be discussed in subsequent chapters. First, it is beneficial to understand the basis of the problems that had to be solved to achieve the objective of long-term storage of cryogens. Storage methods and materials had to be developed that would limit the transfer of heat but that would still allow the container to be used in the intended manner. Therefore, for background information, a summary of the basics of heat transfer and methods that are used to limit heat transfer are given in the following sections. Some thermodynamic principles are discussed in the appendix.

Basic Heat-Transfer Control

Thermodynamics is concerned with energy in various forms and with the transformation of energy from one form to another. Heat is a transient form of energy because it exists only when there is a net exchange of energy between two systems or between a system and its surroundings. A net energy exchange is caused by a temperature difference between the two systems that are involved. In general, this energy transfer is called heat transfer. Heat transfer is a misnomer from a thermodynamic standpoint because the flow of heat is the mechanism of the transfer of internal energy, not the quantity of heat that is transferred. Frequently cryogenics engineers refer to heat transfer as heat leak, another misnomer, but one that is in common use.

Heat-Transfer Modes

Three different modes of heat transfer are known—conduction, convection, and radiation. These three heat-transfer modes have two aspects in common: A temperature difference must exist and the net heat exchange is transferred always in the direction of decreasing temperature. However, the three modes differ completely in the physical mechanisms and laws by which they are governed.

Conduction.—The distinguishing feature of conduction is that the process occurs within the boundaries of a body or across the boundary of one body into another body that is in direct contact with the first body. The mechanism of conductive heat transfer is the exchange of the kinetic energy of motion of the molecules by direct communication and by the drift of free electrons in the case of metals. The fundamental relation for heat transfer by conduction originates from the French physicists Biot and Fourier and is expressed by equation (1).

$$Q_c = kA \frac{\Delta t}{\Delta x} \tag{1}$$

where Q_c = conductive heat-transfer rate, Btu/hr
k = thermal conductivity, Btu/hr-ft °F
A = area, ft²
Δt = temperature difference, °F
Δx = length of the conduction path, ft

The thermal conductivity factor k in equation (1) is a property of the material through which the heat is being transferred.

To limit the amount of heat that is transferred by means of conduction, two major factors have been considered. Methods have been developed to limit or eliminate conduction heat paths such as solid lines, electrical leads, and supports. When conduction heat paths cannot be eliminated, low-thermal-conductivity materials are used to the maximum possible extent.

Convection.—Convection is the process of heat transfer that occurs in fluids (for example, air) because of the mixing of one portion of the fluid with another by means of movements within the fluid. The actual transfer of energy from one fluid particle to another is by means of conduction, but the energy is transported from one point in the system to another by movement of the fluid itself. Equation (2), devised by Newton, defines the fundamental relationship for convective heat transfer.

$$Q_f = hA \ \Delta t \tag{2}$$

where Q_f = convective heat-transfer rate, Btu/hr
h = heat-transfer coefficient, Btu/hr-ft² °F

The heat-transfer coefficient h in equation (2) depends on the composition of the fluid and on the nature and geometry of the fluid motion. Equation (2) means that convective heat transfer occurs at a rate that is proportional to the heat-transfer coefficient, the area of the surface that is in contact with the fluid, and the temperature difference.

The primary device that is used to limit the amount of heat that is transferred to a cryogenic storage vessel by means of convection is an intervening vacuum annulus that surrounds the cryogenic-containing inner vessel. Convective heat transfer cannot occur if there is no fluid to transport the heat. Generally, vacuum levels of approximately 10^{-6} torr are sufficient to limit the heat that is transferred by means of convection so that this heat-transfer method does not contribute significantly to the overall heat transfer.

Radiation.—Thermal radiation is the term that is applied to electromagnetic radiation emitted by a body that has undergone thermal excitation. Any substance at a temperature greater than absolute zero has undergone thermal excitation. When thermal radiation is intercepted by another body, part of the radiation may be transmitted, part may be reflected, or part may be absorbed. The absorbed portion results in heat within the absorbing body. The rate of heat transfer that is caused by radiation between two concentric spheres is defined in equation (3).

$$Q_r = \frac{\sigma A_1 (T_2^4 - T_1^4)}{\dfrac{1}{e_1} + \dfrac{A_1}{A_2}\left(\dfrac{1}{e_2} - 1\right)} \tag{3}$$

where Q_r = radiative heat-transfer rate, Btu/hr
σ = Stefan-Boltzmann constant, Btu/hr-ft² °R⁴
e = emissivity
T = absolute temperature, °R
1 = outer surface of inner sphere
2 = inner surface of outer sphere

Emissivity depends on the type and temperature of the surface, and the subscripts denote the surface that the property describes.

The principal methods for the limitation of heat transfer by means of thermal radiation are radiation shielding and the use of low-emissivity surfaces. Generally radiation shielding in a dewar is accomplished by the placement of one or more concentric surfaces within the vacuum annulus between the pressure vessel (which contains the cryogen) and the outer shell. These shields are nonstructural members, and only intercept and reflect incoming thermal radiation. By means of various manufacturing methods, low-emissivity surfaces (such as silver) are deposited on the outside of the pressure vessel and on the radiation shields. The emissivity of a surface is an indication of the thermal-radiation-reflection efficiency. The lower the emissivity value, the more efficient is the thermal-radiation reflection by the surface. A basic spacecraft dewar cryogenic gas storage system is illustrated in figure 7, and the basic components are identified.

EARLY CRYOGENICS

The beginning of cryogenics can be fixed arbitrarily as Christmas Eve of 1877. Two reports were presented to the French Academy of Sciences in Paris on that day. Both reports concerned the liquefaction of oxygen. One report was presented by an amateur scientist, Louis Cailletet (of Chatillon sur Seine), and the other report was by Raoul Pictet, an engineer (of Geneva). The first successful experiment by Cailletet was performed a few days before Pictet succeeded in liquefying oxygen. However, although Cailletet could claim priority, he could not disclaim the element of luck that was involved. Cailletet profited from an accident that occurred

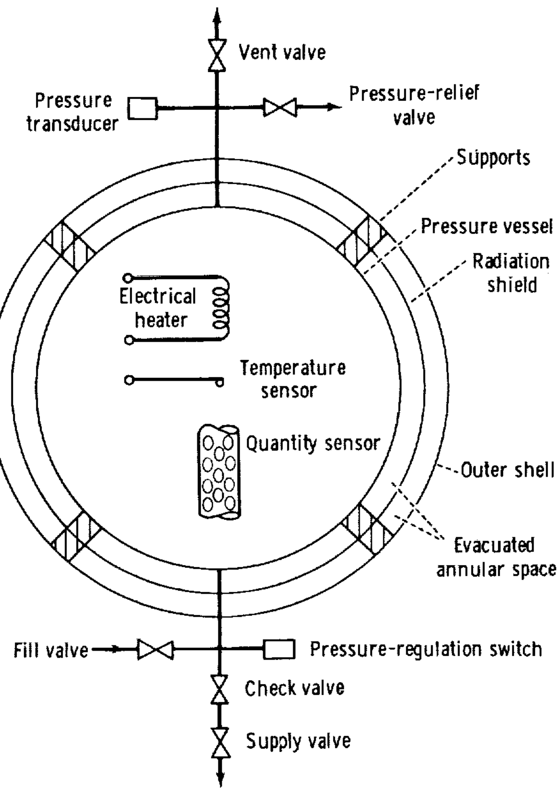


FIGURE 7.—Schematic of a basic spacecraft dewar cryogenic gas storage system.

during one of his early experiments. He was attempting to liquefy acetylene by the application of pressure when a leak developed suddenly in the apparatus. The unintentional rapid expansion of the gas through the hole resulted in liquefaction of the acetylene. Later, Cailletet tried the same high-pressure-expansion procedure on oxygen and succeeded in obtaining small droplets of liquid oxygen (LO_2). It is doubtful that Cailletet realized that he had used a process that has come to be known as Joule-Thomson expansion, a cooling process in which a gas does internal work during expansion.

The method that was used by Pictet was more systematic, and, at least, was indicative of a partial understanding of the significance of the critical point of a gas. For these experiments, Pictet used a mechanical refrigeration cascade system (a series of cooling cycles that involved different fluids that have overlapping liquid regions). The system that was used by Pictet involved boiling sulfur dioxide and carbon dioxide under reduced-pressure conditions.

Although cryogenics began on December 24, 1877, the efforts and lifework of many individuals were necessary before that beginning could be realized. Originally, interest in the liquefaction of gases by several early 19th century chemists and physicists resulted from a desire to investigate the general properties of gases. These men were concerned with the concept of "permanent gases," which involved the belief that certain gases could not be liquefied. The concept was never accepted universally and was discredited each time another "permanent gas" was liquefied. Much of the early work was done by individuals whose training was in other fields, but who demonstrated an understanding of the principles of thermodynamics long before Lord Kelvin, Sadi Carnot, and others reduced thermodynamic principles to mathematical equations and statements. John Gorrie, a Florida physician, was a noteworthy example of such men. In the late 1830's, Gorrie was concerned with the problem of cooling hospital rooms in an effort to alleviate the suffering of malaria patients. He built a machine to make ice, and, in so doing, developed the first expansion engine and counterflow heat exchanger. In later years, these devices were of great value in the field of cryogenics. Most large-scale air-liquefaction systems that are now in use involve the same principle of expanding air through a work-producing device, such as an expansion engine, so that energy can be extracted from the air and the air liquefied. In the early 1820's, Robert Stirling (a Scottish minister) and his brother began to develop a hot-air engine. This development preceded the report on the subject by Carnot in 1824. The Stirling engine was not useful for cryogenic applications in its original form. However, when reversed and used as a heat pump (that is, as a refrigerator), the device became important in the development of cryogenics. In 1860, Kirk (another Scotsman) accomplished the task of reversing the Stirling engine.

Many of the milestones in the development of the field of cryogenics are shown in the following list. Although this list cannot be considered complete, it includes many of the major steps in the evolution of the technology of modern cryogenics.

1820's: Stirling developed a hot-air engine.

1830's: John Gorrie developed the first expansion engine and counterflow heat exchanger.

1853: What is now known as the Joule-Thomson effect was discovered by two English scientists, James P. Joule and William Thomson; Thomson later became Lord Kelvin, after whom the temperature scale is named.

1860: Kirk reversed the Stirling hot-air engine to produce a heat pump.

1877: On December 24, Cailletet and Pictet announced that they had succeeded in the liquefaction of measurable quantities of oxygen.

1883: Wroblevski and Olszewski, Polish physicists, succeeded in the liquefaction of oxygen and nitrogen in appreciable quantities.

1884: Wroblevski produced a mist of liquid hydrogen (LH_2).

1888: D'Arsonval extended the work of Dulong and Petit on heat transfer in rarefied gases and invented vacuum-jacketed glass flasks for the storage of cryogenic fluids.

1892: Sir James Dewar, a Scottish physicist, independently invented and perfected the silvered vacuum flask that now bears his name; the original flasks were made of glass and were coated with mirror silver.

1895: By 1895, Linde and Hampson had discovered a practical use for the Joule-Thomson effect when used in conjunction with a regenerative heat exchanger; they invented a practical device for the liquefaction of air and for the separation of oxygen and nitrogen.

1898: By making use of the availability of large quantities of liquid air, Dewar first liquefied hydrogen in bulk quantity at the Royal Institute of London by means of a combination of the techniques of Pictet and Linde and Hampson.

1907: Linde installed the first air-liquefaction plant in America.

1908: On July 10, Heike Kammerlingh Onnes succeeded in the liquefaction of helium, and confuted the concept of permanent gases; helium was the last gas to be liquefied. This was done by the use of dewar (silvered) flasks, by precooling with liquid hydrogen, and by subsequent Joule-Thomson expansion. Helium for these experiments was obtained by heating monazite sand.

1917: The first natural-gas-liquefaction plant to produce helium was established; the separation of helium from natural gas was one distinctly American achievement.

1926: Goddard test-fired the first cryogenically propelled rocket. Magnetic cooling, the oldest method for the production of temperatures considerably less than 1° K, was suggested first by Debye. Simon introduced the desorption liquefier in 1926 and the single adiabatic-expansion gas liquefier in 1932.

1933: Magnetic cooling was used to attain temperatures less than 1° K. W. F. Giaque (University of California at Berkeley) performed the first successful experiment with a refrigeration cycle that involved adiabatic demagnetization.

1934: Peter Kapitza developed a helium liquefier, which involved the first multiple-expansion engine, to replace the precooling that is supplied ordinarily by liquid hydrogen.

1937: The introduction of evacuated powder insulation resulted in a significant reduction in heat transfer to stored cryogens.

1940: During the early 1940's, development began on the liquid-oxygen converter system.

1942: The German V–2 (V for Vengeance) weapon system was test fired.

1947: S. C. Collins (Massachusetts Institute of Technology) developed the Collins helium cryostat, which consisted of two expansion engines. With associated equipment, these engines had the capability to liquefy helium without precooling with liquid nitrogen.

1952: The National Bureau of Standards (NBS) Cryogenic Laboratory (Boulder, Colorado) was established; this was the first major laboratory established solely for research in cryogenics.

1956: A U.S. Air Force contract was awarded for the development of a liquid-hydrogen-fueled rocket engine.

1957: The Atlas intercontinental ballistic missile (ICBM), propelled by liquid oxygen and RP–1, was test fired.

1958: The development of high-efficiency superinsulation made possible additional reductions in the amount of heat that is transferred to stored cryogens.

1961: The Saturn launch vehicle was test fired; this was the first space vehicle that involved liquid hydrogen and liquid oxygen as a propellant pair.

Some of the basic principles that are involved in understanding the field of cryogenics have been discussed. However, cryogenics is not of interest only in itself. The nature of man requires that cryogenics be applied to uses that will benefit man, and, today, there are many practical applications of cryogenics. Consideration will be given in the following sections to some of the early applications.

THE ROLE OF CRYOGENICS IN EARLY ROCKETRY

The first flight of a liquid-propellant rocket was made on March 16, 1926, on the farm of Miss Effie Ward of Auburn, Mass. This flight was made under the direction of Robert Hutchings Goddard, who is called the father of modern rocketry. As early as 1920, Goddard experimented with liquid-oxygen/gasoline rocket engines. In March 1922, Goddard tested his first liquid-propellant rocket engine. The liquid-propellant rocket was self-pressurizing by means of the expansion of the liquid oxygen and gasoline (in separate tanks); supplementary heat to the fuel tanks was supplied from an alcohol heater. This rocket engine was installed in a rocket body that was built by Goddard and was used to power the history-making flight on March 16, 1926. During this flight, the rocket traveled 184 feet in 2.5 seconds, attaining a speed of approximately 60 miles per hour. Much of the later work by Goddard was done at a test site near Roswell, N. Mex. By 1941, Goddard had developed rockets to a reasonably complete state of operational capability. The devices and concepts that were used in these rockets were similar to those that were used later in the German V–2 weapon system.

German V-2

After the initial research by Goddard, the most significant period in the development of ballistic missiles began in 1942. The first large and practical liquid-propellant rocket, the German V–2, was flight-tested successfully in 1942. This missile was developed by German scientists, including Wernher von Braun, under the direction of Walter Dornberger near the town of Peenemünde on the Baltic Sea. Originally, the V–2 was designated by the Peenemünde staff as the A–4 (A for Aggregate). The V–2 carried 8000 pounds of alcohol as fuel and 11 000 pounds of liquid oxygen to support combustion of the fuel. It has been estimated that during 1944 and 1945, the Germans produced several thousand V–2 rockets, and that they launched 4300 of them at Allied targets. The configuration of the V–2 was representative of many rockets that were developed later. The alcohol and liquid oxygen were stored in separate tanks and were piped to the rocket-engine combustion chamber. If Goddard is considered to be the father of modern rocketry, surely von Braun must be considered its principle champion. Von Braun, who began to emphasize rocket research in 1942, has been instrumental in rocket research since that time and is still a leader in the field of rocketry.

Ballistic Missiles

The U.S. Army ballistic-missile program was started after World War II but was canceled in 1947 for economic reasons. Interest in the program continued, and another contract was initiated in 1951 that resulted in the establishment of the Atlas program. By 1955, major decisions were made that completed the design concept and established the final configuration of the Atlas and the Titan intercontinental ballistic missiles. The purpose of these two systems was to deliver a multimegaton warhead accurately onto a military target 6000 miles away. The Atlas and Titan systems and the Thor and Jupiter systems (intermediate range missiles) became the mainstays for the first several years of space exploration. These four systems have similar propulsion systems and involve liquid oxygen as a propellant oxidizer. Some basic information on the V–2, the Titan ICBM, and the Saturn V is given in table 3. The table is illustrative of some of the progress that has been achieved in cryogenic propulsion systems.

An analysis of propellant performance and an assessment of logistics problems were conducted in 1954 in support of the ICBM program. The result of this effort was the determination that the cryogenic-propellant combination of liquid oxygen and RP–1 was the best and most logical choice. This combination generated enough energy to meet the range and payload requirements. Consequently the logistics problems could be solved. This combination was the only propellant pair that was capable of meeting the mission requirements within the short time schedule that was imposed on the program because of the need to establish a deterrent to possible attack. Later, when more time and an adequate supply of hydrogen were available, a liquid-oxygen/liquid-hydrogen system was developed. Currently this type of system is being used in the upper stages of such launch vehicles as the Saturn V. To implement the ICBM program, it was necessary to develop rapidly such items as large ground-storage dewars, propellant-loading and propellant-handling techniques, thermal insulation, cryogenic hardware components, and high-pressure low-temperature gas technology. Initially the basic V–2 cryogenics data were used because the data constituted the sole candidate for consideration at the time.

TABLE 3.—*Basic Characteristics of the German V–2, the Titan ICBM, and the Saturn V Launch Vehicles*

Characteristic	Launch vehicle		
	V–2	Titan	Saturn V
Propulsion type			
First stage	LO_2/alcohol	LO_2/RP–1	LO_2/RP–1
Second stage	None	LO_2/RP–1	LO_2/LH_2
Third stage	None	None	LO_2/LH_2
Thrust, lb			
First stage	55 100	300 000	7 570 000
Second stage	None	80 000	1 125 000
Third stage	None	None	225 000
Nominal speed, mph	3600	17 000	[a] 17 446 [b] 24 188
Overall height, ft	46	97	363
Launch weight, lb	28 400	220 000	6 034 000
First system test flight	1942	Feb. 6, 1959	[c] Nov. 9, 1967

[a] At orbital insertion
[b] At translunar injection
[c] Apollo 4

THE DEVELOPMENT OF LIQUID-OXYGEN CONVERTERS FOR AIRCRAFT

Oxygen is required by man and must be carried in any manned vehicle if oxygen is not present in sufficient concentration in the external environment. The methods that are available for oxygen storage are ambient-temperature storage, cryogenic storage, and chemical storage. Cryogenic storage has been favored for use in military aircraft since the early 1940's. At that time, the need to increase the operational range of military aircraft resulted in the need to increase the quantity of oxygen carried. A major design constraint was that the space available for a new system was to be limited to that space occupied by the low-pressure oxygen-gas cylinders that were being used. To increase the quantity of oxygen available without increasing the space occupied meant that high-pressure gas storage and cryogenic gas storage were the only suitable methods. Subsequently, the liquid-oxygen converter system was developed to meet these requirements. The use of a liquid-oxygen system instead of a gaseous-oxygen system made possible a reduction in weight, volume, and maintenance of approximately 75 percent.

The proposed use of liquid oxygen as a source of breathable oxygen in aircraft was not a new idea. As early as 1921, work had been started on this system. However, it was not until the 1940's that a practical system for the conversion of liquid oxygen to a gas suitable for breathing was proposed by the National Bureau of Standards. The proposed system did not involve an external source of energy other than the surrounding atmosphere for the conversion of liquid oxygen to breathable gas. Basically this is the same type of system that is in use today.

A component schematic of a liquid-oxygen-converter system is shown in figure 8. The basic system consists of a dewar-type, vacuum-insulated container. Because of space limitations, the conventional vacuum-insulated container, which has a long, low-heat-conducting neck, could not be used (fig. 9).

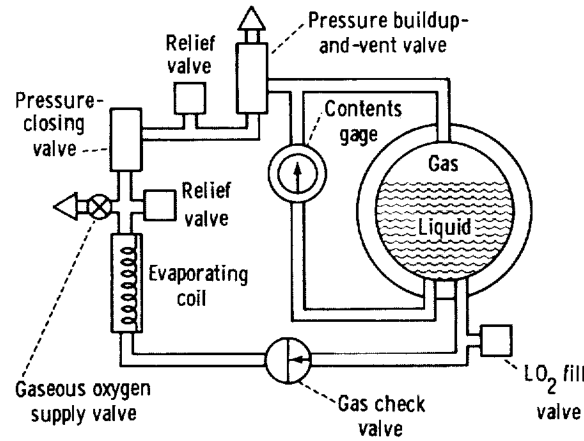


FIGURE 8.—Component schematic of a liquid-oxygen-converter system.

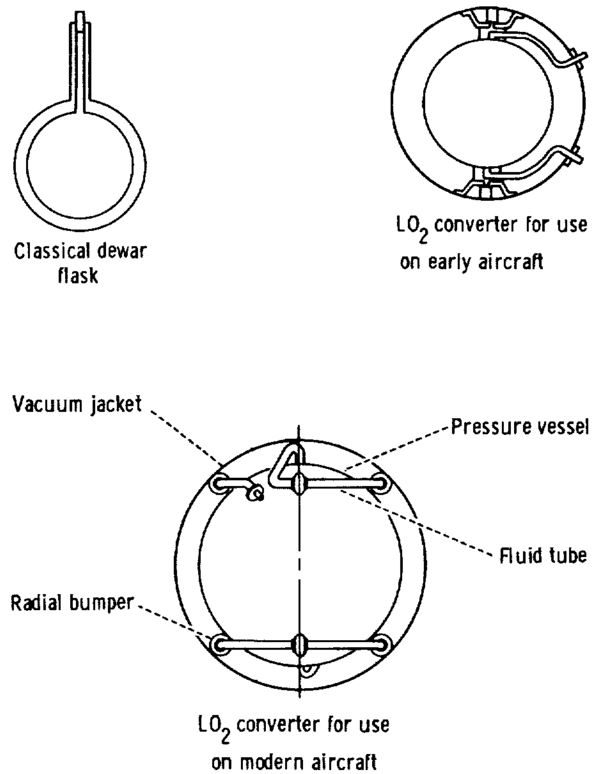


FIGURE 9.—Schematic of the classical dewar flask and two liquid-oxygen-converter systems.

A container that would contain the required quantity of oxygen but that did not have a long neck had to be developed. The container that holds the liquid oxygen is enclosed completely by a second concentric container. All air between the two containers is evacuated so that convective heat transfer is minimized. The inner container, which contains the liquid oxygen, is connected to the outer container by means of two spiral tubes; these tubes are the fill and supply lines. They are made long to provide a long heat path between the outer and inner containers, thereby reducing conductive heat transfer. Low-thermal-conductivity members are used to support the inner container within the outer container. When installed in an aircraft, the pressure buildup-and-vent valve and the filler valve are located side by side behind a hatch in the skin of the aircraft. This placement of the valves makes them accessible for service purposes. Simplified schematics of a classical dewar flask, an early liquid-oxygen converter, and a modern liquid-oxygen converter are shown in figure 9. The conventional vacuum-insulated container is shown for comparative purposes.

On military aircraft, liquid-oxygen converters are now standard equipment as a source of breathable oxygen at high altitudes. Liquid-oxygen-converter systems that have a capacity of 10 liters can supply sufficient oxygen at the correct temperature and pressure to maintain three men at high altitudes for approximately 2 hours. Such systems are approximately 12 inches in height and 13 inches in diameter.

The present aerospace cryogenic gas storage and supply systems technology evolved with the development and application of the liquid-oxygen converter. The major technological developments that accompanied aircraft liquid-oxygen converter evolution were vacuum integrity; low-thermal-conductivity annular suspension systems; high-structural-integrity tank, joint, and suspension-system designs; and low-emissivity radiation surfaces.

2 Space Applications

The purpose of the cryogenic gas storage system (CGSS) is to store and supply certain fluids for the following functions:

(1) Oxygen: Environmental control system (ECS), electrical power system (EPS) (fuel cells)

(2) Nitrogen: Environmental control system

(3) Hydrogen: Electrical power system (fuel cells)

(4) Helium: Pressurization of propellants

To date, all of the spacecraft cryogens except propellants have been stored in double-wall containers known as dewars. However, a CGSS that does not involve the dewar concept has been developed and tested; this system is known as a single-wall tank. Unlike a dewar system, the pressure vessel is insulated externally and a vacuum annulus is not used. The single-wall tank is discussed in another section of this chapter.

Cryogenic gas storage systems have been developed for the storage of liquid oxygen (LO_2), liquid hydrogen (LH_2), liquid nitrogen (LN_2), and liquid helium (LHe). In this report, cryogenic gas storage system will mean those cryogenic gas storage systems that are used for purposes other than primary vehicle propulsion (for example, Saturn V). Cryogenic storage is useful because of the high liquid density of the cryogens and the low storage pressure. These characteristics result in smaller container sizes, lower container-strength-requirement levels, and lower tankage weights. The CGSS has been used effectively as standard equipment on the Gemini and Apollo spacecraft. The Mercury spacecraft did not involve a CGSS because of the minimal requirement for metabolic oxygen and because of no requirement for electrical power system cryogens. A high-pressure tank was used to supply metabolic oxygen for the astronaut, and batteries were the electrical power source for the spacecraft. In addition to the Gemini and Apollo cryogenic gas storage systems, other systems have been developed and tested in the course of advancing the state of the art. All of the cryogenic gas storage systems that have been developed, tested, and used in manned spacecraft are described in this chapter.

SYSTEM CONSIDERATIONS OF SPACECRAFT CRYOGENIC GAS STORAGE SYSTEMS

Cryogenic gas storage systems have been used successfully in the Gemini and Apollo Programs. Both the Gemini and Apollo spacecraft carried liquid oxygen and liquid hydrogen, which were expelled as required to the electrical power (fuel cells) and environmental control systems. The Gemini and Apollo space flights have proven the feasibility of the concept and the system reliability of the cryogenic gas storage systems that were developed for use in these programs. Progress from the aircraft liquid-oxygen converters to the Gemini CGSS to the Apollo system was necessary for the establishment of the manufacturing technology that was required for the production of these systems. Many contractors and thousands of individuals participated in and contributed to the development of the technology that was required for the Gemini and Apollo systems.

This development was required in many scientific disciplines and for many of the components that make up a CGSS. A CGSS is composed of several internal and external components. The internal components of a dewar system generally include temperature, pressure, and quantity sensors, and motor fans and heaters for fluid expulsion. External dewar-system components include the mounting structure, plumbing, valves, filters, signal conditioners, switches, transducers, regulators, heat exchangers, and the associated wiring and connectors. Also, dewar systems may include an ion pump, which is used to maintain a vacuum within the annulus between the concentric walls. A functional description of the major components just mentioned is included in the last section of this chapter.

The design of a spacecraft CGSS is different from the design of a terrestrial CGSS, primarily because of the zero-gravity environment of space, the adverse operational environment, and the inaccessibility for repair or modification during operation. Cryogen thermal stratification and possible random orientation of the liquid and vapor phases require appropriate design consideration for the expulsion and quantity measurement of the stored fluid. A major consideration in the design of any CGSS is the storage technique to be used. Each of these factors must be considered in conjunction with system/mission design considerations (such as the required fluid quantity) so that an optimum system is the result.

Specific system/mission considerations are given in the following list.

(1) Mission considerations

(a) Reliability: The reliability for any system on manned spacecraft is of paramount importance.

(b) Operational pressure: The operational pressure may be greater or less than the critical pressure of the fluid; hence, cryogens may be stored in the single-phase (supercritical) or two-phase (subcritical) condition. A discussion of both of these storage techniques and the associated operational characteristics is given in the next section.

(c) Quantity measurement: The quantity-measurement accuracy that is required may dictate the thermodynamic design. A supercritical (homogeneous fluid) system involves a simple and effective method of quantity measurement, whereas a subcritical (liquid and vapor mixture) system involves a more complex and possibly a less accurate quantity-measurement method.

(d) Pressure control: The extent of pressure control that is required may influence the thermodynamic design. A subcritical system may undergo pressure instabilities that are caused by random liquid-vapor orientation. A supercritical system may undergo pressure drops that are caused by thermal stratification and subsequent fluid mixing.

(2) Manufacturing considerations

(a) Reproducibility: Predictable and consistent performance of a CGSS is critical to the success of any mission.

(b) Shelf life: A CGSS must be designed and fabricated to withstand a shelf life of as many as 3 years without any serious degradation.

(c) Weight: As for most other systems on a spacecraft, the CGSS must be designed for minimum weight.

(d) Materials: The materials that are selected must be compatible with the environment and the cryogens and must have high strength-to-weight ratios.

(e) Envelope constraints: Spacecraft-interfacing constraints may influence the physical and structural design parameters.

(3) Performance considerations

(a) Standby time: The dormant period between filling and launch may influence the fill quantity and method of storage (subcritical or supercritical).

(b) Fluid quantity: The CGSS size, weight, and capacity are constrained by the quantity of usable fluid, the residual fluid quantity, the ullage, and the contingency requirements.

(c) Fluid-usage rate: The fluid-flow rate will dictate the size, design, and power requirements of the heater.

(d) Power requirements: Power requirements for the CGSS controls, instrumentation, motor fans, and heaters should be compatible with the availability and type of spacecraft power.

(e) Environment conditions: Environmental temperatures will influence significantly the thermal and thermodynamic design. A high environmental temperature would necessitate the use of better insulation and, possibly, vapor cooling of the insulation during fluid withdrawal. Also, consideration must be given to the possibility of a necessity to vent fluid overboard, making a larger storage system necessary.

Storage-Technique Considerations

The three most common methods of gas storage for spacecraft application are high-pressure gas storage at ambient temperature, single-phase cryogenic-fluid storage at supercritical pressure, and two-phase cryogenic-fluid storage at subcritical pressure.

High-pressure storage. — High-pressure gas-storage systems are simple, reliable, and have indefinite standby capability. However, the large volume and weight that are required even for a small payload are serious disadvantages, as can be seen in figures 10 and 11.

Supercritical storage.—Supercritical storage means that the cryogenic fluid is stored at a pressure greater than the critical pressure of the fluid and that the fluid exists in a single phase throughout the duration of the mission. A pressure-enthalpy diagram of the supercritical storage of cryogenic fluids is given in figure 12. Point 1 is indicative of the initial fill condition, which is a mixture of saturated

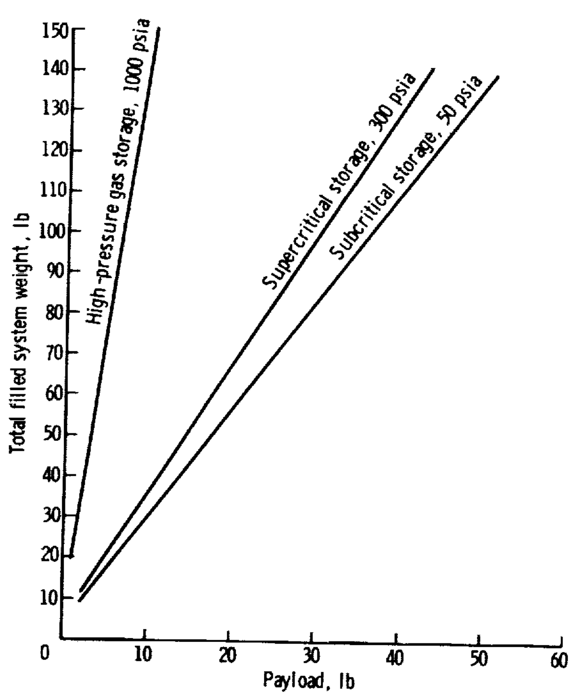


FIGURE 10.—Total hydrogen tankage weight plotted as a function of payload.

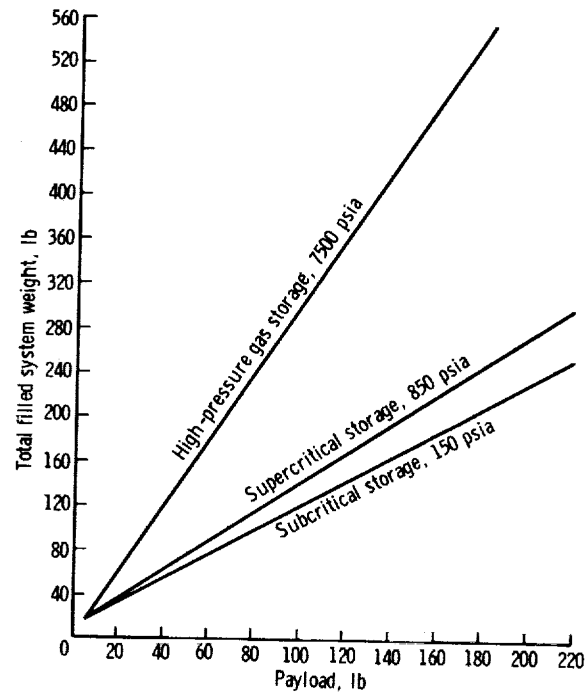


FIGURE 11.—Total oxygen tankage weight plotted as a function of payload.

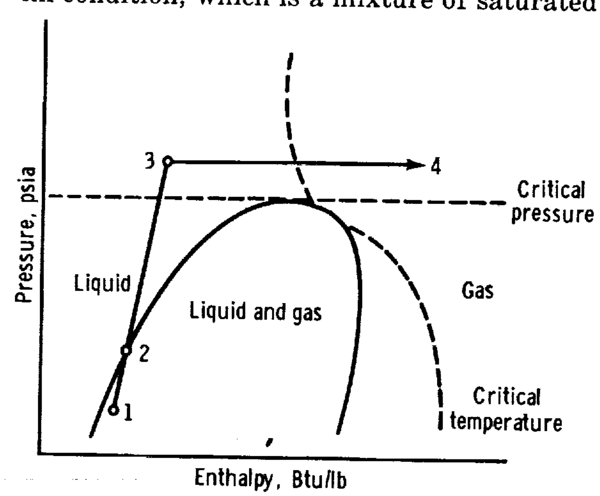


FIGURE 12.—Pressure-enthalpy diagram of supercritical storage.

liquid and vapor at atmospheric pressure. After fill, heat input causes the pressure to increase at a constant density (density based on vessel volume). During the process from point 1 to point 2, the liquid expands and the ullage gas condenses until the entire volume is filled with liquid. From point 2 to point 3, thermal energy is absorbed by the stored fluid, and the fluid pressure is increased above the critical pressure. Constant supercritical operation, path 3 to 4, is achieved by heating the stored fluid during fluid withdrawal. During supercritical operation, the stored mass remains a homogeneous single-phase fluid, and fluid expulsion is assured because of the high pressure.

Operation of a typical supercritical system (fig. 13) starts with the purging of the system internal vessel and fluid lines. Usually purging is accomplished by means of vacuum pumping, which ensures that static and dead-ended parts are free from contamination. After the system has been purged, the vessel is filled with a saturated liquid cryogen at ambient pressure. Initially most of the liquid is vaporized as it enters the pressure vessel, and vapor is vented through the vent valve. The fill line enters the pressure vessel at the bottom of the vessel, and the vent line exits the pressure vessel close to the top at a point that corresponds to the design fill level. As the temperature of the pressure vessel decreases, the liquid level rises. Eventually the system is

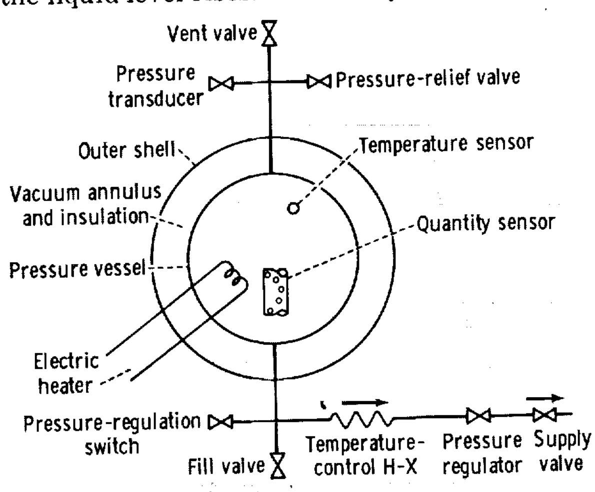


FIGURE 13.—Supercritical CGSS.

chilled to an equilibrium temperature by allowing it to continue to vent boil-off vapors through the fluid-vent line. After chilldown, the vessel is topped-off to capacity with additional saturated liquid cryogen. Then the fill-and-vent valves are closed, and the fluid pressure begins to increase because of liquid expansion that is caused by the energy absorbed from heat that is entering the system. Eventually the liquid phase completely fills the inner vessel; the fluid is in the compressed-liquid region. If a more rapid pressure rise is desired, an electrical heater inside the vessel may be energized. After the required pressure (which is above the critical pressure) is reached, the system supply valve is opened, and the fluid flows out of the vessel through an external temperature-control heat exchanger (H–X) and pressure regulator. As the fluid flows out of the storage system, the fluid pressure decreases. However, an electrical pressure-regulation switch operates automatically to energize the internal heater whenever the pressure decays below a predetermined level. Thus the thermal energy that is absorbed by the stored fluid maintains supercritical pressure operation.

Subcritical storage.—Subcritical storage means that the cryogenic fluid is stored at a pressure less than the critical pressure of the fluid and that the cryogen exists as a liquid-vapor mixture. A pressure-enthalpy diagram of the subcritical cryogenic storage is given in figure 14. Point 1 is indicative of the initial fill condition, which is a mixture of saturated liquid and vapor at atmospheric pressure. After the vessel is filled, heat is added to raise the fluid pressure to the operating pressure (point 3). Constant operating pressure, point 3 to point 4, is maintained by vaporizing the stored liquid during liquid withdrawal.

Operation of a typical subcritical system is similar to operation of a supercritical system for the purge, fill, chilldown, top-off, and heater operation for initial pressurization. Delivery techniques, however, involve a consideration of both the liquid and gas phases

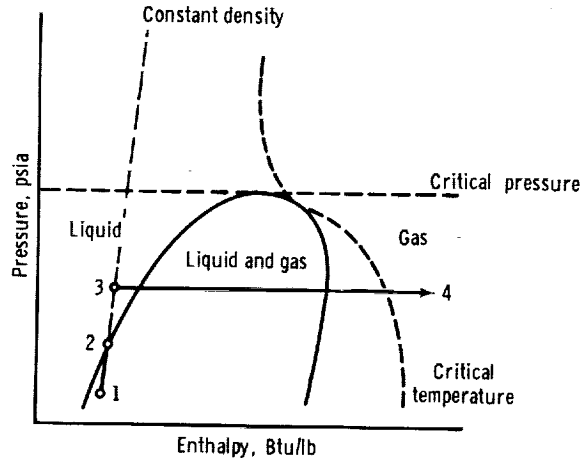


FIGURE 14.—Pressure-enthalpy diagram of subcritical storage.

that are present in a subcritical system. A subcritical system must be designed so that either a liquid or a vapor may be withdrawn from the pressure vessel and so that the liquid or gas may be conditioned to the required density, pressure, and temperature before it leaves the system. If liquid is withdrawn, it must be converted to a vapor by means of heat exchangers. A subcritical storage system that involves heat exchangers for expelled-vapor conditioning is shown in figure 15. A typical subcritical system is equipped with an internal regulator valve that throttles the

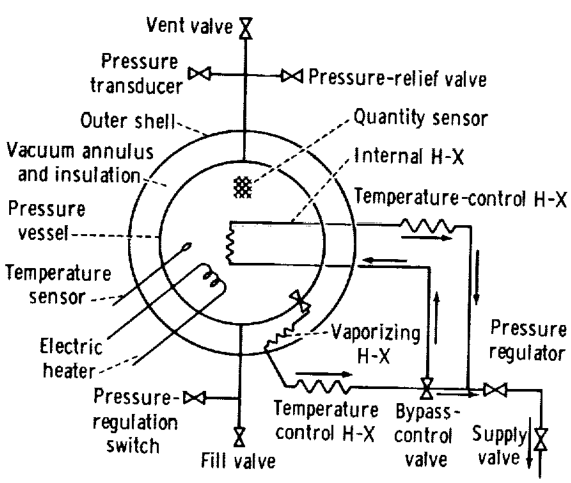


FIGURE 15.—Subcritical CGSS.

fluid being withdrawn from the pressure vessel to a predetermined delivery pressure. The throttling process causes the fluid to expand and its temperature to decrease to a point below the temperature of the stored liquid. Then the liquid leaves the internal regulator valve, passes through a heat exchanger, and vaporizes prior to exiting the dewar. If required, the external temperature-control heat exchanger transmits additional heat to the vapor. As the vapor flows from the system, the fluid pressure decreases; however, a pressure-bypass control valve operates automatically to cause the warm gas to flow into the internal heat exchanger, wherein heat is transferred to the cold liquid in the pressure vessel. The thermal energy that is absorbed by the liquid in the pressure vessel maintains the constant operating pressure. After leaving the internal heat exchanger, the vapor is reheated in the second external heat exchanger before passing through the pressure regulator and out of the system. When the tank pressure is equal to or greater than the operating pressure, the flow is directed to the regulator without passage through the internal heat exchanger.

Gemini Cryogenic Gas Storage System

On January 3, 1962, the Gemini Program was recognized officially. The Gemini two-man spacecraft served as an intermediate step between Project Mercury and the Apollo Program and had the following prime objectives.

(1) Expose the astronauts and supporting equipment to long-duration flights

(2) Rendezvous and dock with another orbiting vehicle

(3) Experiment with extravehicular activity (EVA) while in orbit

The first space-flight CGSS was used on the Gemini spacecraft for the storage and expulsion of supercritical oxygen and hydrogen. Oxygen and hydrogen were used by the fuel cells, which served as the primary source of electrical power for the spacecraft. A fuel cell is an electrochemical device that converts the energy generated when hydrogen and oxygen are combined electrocatalytically into direct current in an external circuit. Water is produced as a byproduct of the process. Oxygen was supplied to the ECS, which supplies an oxygen environment and controls cabin pressurization, flow rate, suit cooling, humidity, and purification of the atmosphere.

The Gemini CGSS consisted of six different types of tanks that were designed based on the requirements of both the 2-day and 14-day missions. The oxygen required for the ECS and the fuel cell reactant supply system (RSS) was supplied from two separate tanks (table 4). For thermal protection of the stored fluids, the Gemini dewar system was equipped with aluminized Mylar insulation within the vacuum annulus as a radiant-heat barrier. The pressure vessel was supported within the outer shell by means of Fiberglas pads. The fluid-equilibration heater system consisted of perforated copper spheres that had electric heater elements coiled and fastened to the external surface of the copper spheres. A typical Gemini cryogenic gas storage tank, tank system, and spacecraft RSS are shown in figures 16 to 18. The CGSS design characteristics for the six different tanks are given in tables 5 to 7. Actual storage-tank thermal-performance data are shown in table 8. The Gemini CGSS was furnished by a contractor under contract to the Gemini spacecraft manufacturer.

TABLE 4.—*Lifetime Ratings of the Six Gemini CGSS Tanks*

Cryogen	ECS tank lifetimes, days	EPS/RSS tank lifetimes, days
Oxygen [a]	2	2
	14	14
Hydrogen	None	2
		14

[a] Although the lifetime ratings for the ECS and EPS/RSS O_2 tanks are identical, the tank capacities are different.

Supercritical storage of the required cryogens was chosen for the Gemini spacecraft because of the problems (such as fluid orientation and quantity measurement) that were associated with subcritical storage in zero gravity and because of increased fluid-quantity requirements. A discussion of some of the problems incurred by subcritical storage in a zero-gravity environment is contained in a subsequent section. The Gemini fluid-quantity requirements were greater than the fluid-quantity requirements for the Mercury spacecraft because of the following reasons.

(1) The use of fuel cells as the electrical power source necessitated the use of hydrogen in addition to more oxygen.

(2) The extended mission duration (14 days).

(3) The larger crew (two men), necessitating more metabolic oxygen.

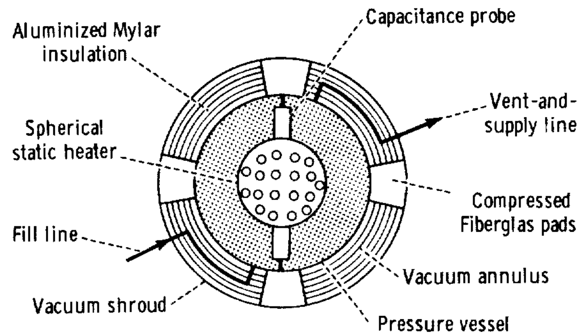


FIGURE 16.—Typical Gemini cryogenic storage tank.

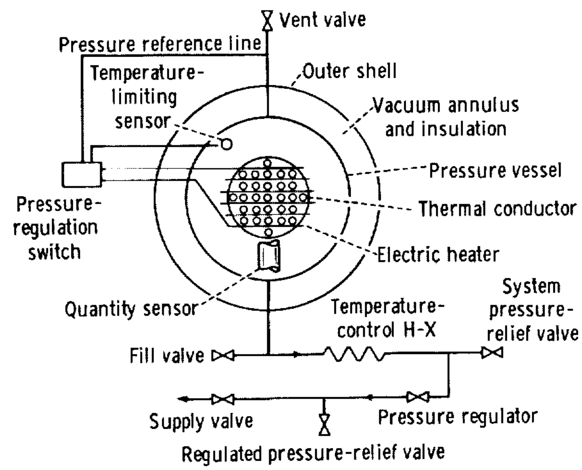


FIGURE 17.—Typical Gemini CGSS.

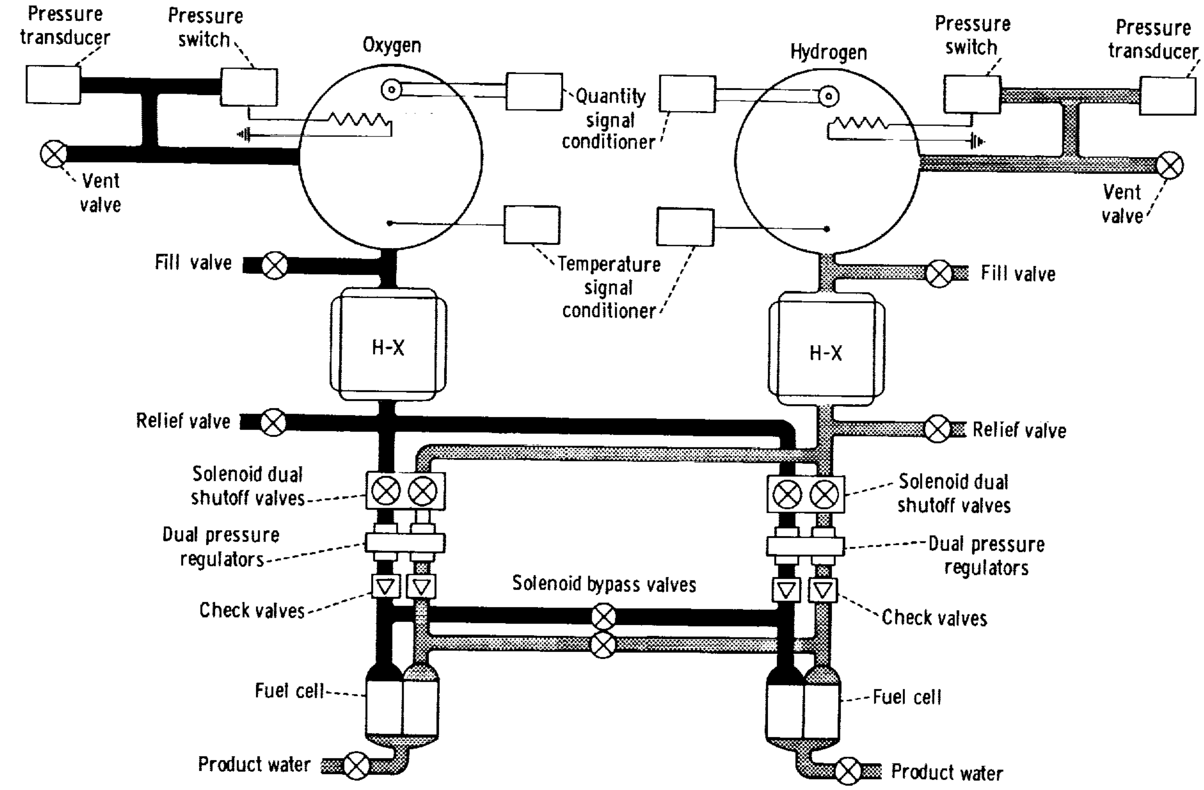


FIGURE 18.—The Gemini RSS.

TABLE 5.—*Gemini CGSS Design Characteristics (ECS)*

Gemini ECS supercritical oxygen-storage system	2-day ECS oxygen tank	14-day ECS oxygen tank
Mission time, days	2	14
Dry system weight, ±5 percent, lb	16.30	41.79
Usable fluid per vessel, lb	15.3	104.0
Fluid weight at fill, minimum, lb	15.75	106.0
Ullage, percent	15.4	9.7
Normal operating pressure, psig	$850 \begin{smallmatrix}+60\\-50\end{smallmatrix}$	$850 \begin{smallmatrix}+60\\-50\end{smallmatrix}$
Maximum operating pressure		
At 70° F, psig	930	930
At −160° F, psig	1000	1000
Relief-valve pressure range, psia	$1000 \begin{smallmatrix}+0\\-55\end{smallmatrix}$	$1000 \begin{smallmatrix}+0\\-55\end{smallmatrix}$
Proof pressure		
At 70° F, psig	1550	1550
At −160° F, psig	1670	1670
Burst pressure		
At 70° F, psig	2000	2000
At −160° F, psig	2220	2220
Minimum flow rate at 100° F environment, lb/hr	0.286	0.286
Maximum flow rate, lb/min	0.35	0.35
Design standby time, hr	72	72
Revised standby time,[a] hr	50	96
Fluid vented during standby, lb	None	None
Design heat leak,[a] Btu/hr	5.5	10.55
Internal temperature, °F	−280	−162

TABLE 5.—*Gemini CGSS Design Characteristics (ECS)—Concluded*

Gemini ECS Supercritical oxygen-storage system	2-day ECS oxygen tank	14-day ECS with copper
Environmental temperature, °F	160	100
Off-design heat leak,[a] Btu/hr	—	17.8
Internal temperature, °F	—	−280
Environment temperature, °F	—	160
Internal heater operation, Vdc	28	28
Normal: Automatic pressure control, W	12±2	12±2
Emergency: Manual control, W	$550\ ^{+25}_{-0}$	$325\ ^{+25}_{-0}$
Destratification device	Static thermal with copper conductor	Static thermal with copper conductor
Pressure vessels		
Volume, minimum, ft^3	0.262	1.65
Inside diameter, in.	9.562	17.60

Gemini ECS Supercritical oxygen-storage system	2-day ECS oxygen tank	14-day ECS oxygen tank
Material	Inconel 718	Inconel 718
Wall thickness, nominal, in.	0.032	0.058
Outer shells		
Inside diameter, in.	12.08	19.95
Wall thickness, nominal, in.	0.0225	0.027
Material	Ti–5Al–2.5Sn	Ti–5Al–2.5Sn
Collapse pressure, psia	20	20
Insulation, aluminized Mylar		
Layers per vessel	65	65
Thermal conductivity, Btu/ft-hr-°R	$1\ to\ 3\times10^{-4}$	$1\ to\ 3\times10^{-4}$
Pressure-vessel support pads, Fiberglas		
Number per vessel	4	12
Thermal conductivity, Btu/ft-hr-°R	$6\ to\ 7\times10^{-4}$	$6\ to\ 7\times10^{-4}$
Density, lb/ft^3	20	20
Total area, ft^2	0.436	2.03
Pad diameter, in.	2.0	2.0

[a] Per McDonnell Aircraft Corporation

TABLE 6.—*Gemini CGSS Design Characteristics (RSS, Oxygen)*

Gemini RSS supercritical oxygen-storage system	2-day RSS oxygen tank	14-day RSS oxygen tank
Mission time, days	2	14
Dry system weight, ±5 percent, lb	23.61	59.06
Usable fluid per vessel, lb	45.0	177.4
Fluid weight at fill, minimum, lb	46.0	180.0
Ullage, percent	10.4	4.0
Normal operating pressure, psig	$850\ ^{+60}_{-50}$	$850\ ^{+60}_{-50}$
Maximum operating pressure		
At 70° F, psig	930	930
At −160° F, psig	1000	1000
Relief-valve pressure range, psig	$1000\ ^{+0}_{-55}$	$1000\ ^{+0}_{-55}$

Gemini RSS supercritical oxygen-storage system	2-day RSS oxygen tank	14-day RSS oxygen tank
Proof pressure		
At 70° F, psig	1550	1550
At −160° F, psig	1670	1670
Burst pressure		
At 70° F, psig	2000	2000
At −160° F, psig	2220	2220
Minimum flow rate		
At 160° F, lb/hr	—	0.490
At −160° F, lb/hr	0.540	—
Maximum flow rate at −60° F, lb/hr	2.22	2.22
Design standby time, hr	72	72
Revised standby time,[a] hr	65	55
Fluid vented during standby, lb	None	None

TABLE 6.—*Gemini CGSS Design Characteristics (RSS, Oxygen)—Concluded*

Gemini RSS supercritical oxygen-storage system	2-day RSS oxygen tank	14-day RSS oxygen tank
Design heat leak,[a]		
Btu/hr	10.8	20.9
Internal temperature, °F	−280	−280
Environmental temperature, °F	160	160
Off-design heat leak,[a]		
Btu/hr	—	12.5
Internal temperature, °F	—	−162
Environmental temperature, °F	—	100
Internal heater operation, Vdc	28	28
Normal: Automatic pressure control, W	$78\ {}^{+0}_{-16}$	$78\ {}^{+0}_{-16}$
Emergency: Manual control, W	None	None
Destratification device	Static thermal with copper conductor	Static thermal with copper conductor
Pressure vessels		
Volume, minimum, ft^3	0.721	2.64
Inside diameter, in.	13.35	20.56
Material	Inconel 718	Inconel 718
Wall thickness, nominal, in.	0.0455	0.070
Outer shells		
Inside diameter, in.	15.56	22.942
Wall thickness, nominal, in.	0.0225	0.0325
Material	Ti–5Al–2.5Sn	Ti–5Al–2.5Sn
Collapse pressure, psia	20	20
Insulation, aluminized Mylar		
Layers per vessel	65	65
Thermal conductivity, Btu/ft-hr-°R	$1\ to\ 3\times10^{-4}$	$1\ to\ 3\times10^{-4}$
Pressure-vessel support pads, Fiberglas		
Number per vessel	6	12
Thermal conductivity, Btu/ft-hr-°R	$6\ to\ 7\times10^{-4}$	$6\ to\ 7\times10^{-4}$
Density, lb/ft^3	20	20
Total area, ft^2	0.904	3.37
Pad diameter, in.	2.0	2.0

[a] Per McDonnell Aircraft Corporation

TABLE 7.—*Gemini CGSS Design Characteristics (RSS, Hydrogen)*

Gemini RSS supercritical oxygen-storage system	2-day RSS oxygen tank	14-day RSS oxygen tank
Mission time, days	2	14
Dry system weight, ±5 percent, lb	27.52	50.41
Usable fluid per vessel, lb	5.6	21.9
Fluid weight at fill, minimum, lb	5.80	22.25
Ullage, percent	33.1	8.0
Normal operating pressure, psig	$250\ {}^{+0}_{-40}$	$250\ {}^{+0}_{-40}$
Maximum operating pressure		
At 70° F, psig	200	200
At −320° F, psig	350	350

Gemini RSS supercritical hydrogen-storage system	2-day RSS hydrogen tank	14-day RSS hydrogen tank
Relief-valve pressure range, psig	$350\ {}^{+0}_{-35}$	$350\ {}^{+0}_{-35}$
Proof pressure		
At 70° F, psig	335	335
At −320° F, psig	585	585
Burst pressure		
At 70° F, psig	440	440
At −320° F, psig	777	777
Minimum flow rate at 160° F environment, lb/hr	0.070	0.064
Maximum flow rate at −60° F environment, lb/hr	0.276	0.276

TABLE 7.—*Gemini CGSS Design Characteristics (RSS, Hydrogen)—Concluded*

Gemini RSS supercritical hydrogen-storage system	2-day RSS hydrogen tank	14-day RSS hydrogen tank
Design standby time, hr	72	72
Fluid vented during standby, lb	None	None
Design heat leak,[a] Btu/hr	7.26	7.25
Internal temperature, °F	−415	−392
Environment temperature, °F	160	100
Internal heater operation, Vdc	28	28
Normal: Automatic pressure control, W	18±2	18±2
Emergency: Manual control, W	None	None
Destratification device	Static thermal with copper conductor	Static thermal with copper conductor
Pressure vessels		
Volume, minimum, ft³	1.97	5.49
Inside diameter, in.	18.65	26.25
Material	Ti–5Al–2.5Sn	Ti–5Al–2.5Sn
Wall thickness, nominal, in.	0.030	0.056
Outer shells		
Inside diameter, in.	21.466	28.928
Wall thickness, nominal, in.	0.0305	0.040
Material	Ti–5Al–2.5Sn	Ti–5Al–2.5Sn
Collapse pressure, psia	20	20
Insulation, aluminized Mylar		
Layers per vessel	65	65
Thermal conductivity, Btu/ft–hr–°R	1 to 3×10^{-4}	1 to 3×10^{-4}
Pressure-vessel support pads, Fiberglas		
Number per vessel	6	10
Thermal conductivity, BTU/ft–hr–°R	6 to 7×10^{-4}	6 to 7×10^{-4}
Density, lb/ft³	20	20
Total area, ft²	0.294	0.775
Pad diameter, in.	2.0	2.0

[a] Per McDonnell Aircraft Corporation

TABLE 8.—*Gemini CGSS Thermal-Performance Data*

Manufacturer serial no.	Vent heat loss, lb/hr at 75° F ambient	Fluid	Vented heat leak, Btu/hr	Nonvented heat leak, Btu/hr	Standby time, hr at 75° F environment	
					Predicted	Actual
2-day ECS oxygen vessel [a]						
2	0.0635	O_2	5.82	6.05	65.0	>72
3	0.0484	O_2	4.44	4.50	88.2	>72
5	0.0834	O_2	7.63	8.05	49.1	48.8
6	0.077	N_2	6.57	6.85	57.4	>72
12	0.110	N_2	9.40	10.00	39.0	(b)
14	0.084	N_2	7.20	7.55	52.0	(b)
14-day ECS oxygen vessel [c]						
2	0.154	N_2	13.1	13.6	128.8	>72
9	0.1502	O_2	13.8	14.65	119.0	>72
10	0.194	N_2	16.5	18.65	96.0	>72
16	0.195	N_2	16.6	18.80	96.5	(b)
2-day RSS oxygen vessel [d]						
2	0.11	N_2	9.39	10.25	80.3	(b)
3	0.1466	O_2	13.44	14.6	55.5	51
6	0.1175	N_2	10.0	10.9	75.0	>72

TABLE 8.—*Gemini CGSS Thermal-Performance Data—Concluded*

Manufacturer serial no.	Vent heat loss, lb/hr at 75° F ambient	Fluid	Vented heat leak, Btu/hr	Nonvented heat leak, Btu/hr	Standby time, hr at 75° F environment	
					Predicted	Actual
			2-day RSS oxygen vessel [d]			
8	0.113	N_2	9.63	10.5	78.0	69
9	0.105	N_2	8.97	9.75	84.5	68.9
10	0.1345	N_2	11.5	12.5	65.3	66
11	0.118	N_2	10.05	10.95	75.0	(b)
12	0.1191	N_2	10.12	11.05	74.0	>72
13	0.1245	N_2	10.65	11.55	70.5	(b)
15	0.1275	N_2	10.84	11.80	69.0	67
17	0.114	N_2	9.71	10.60	77.3	(b)
			14-day RSS oxygen vessel [e]			
2	0.199	N_2	17.0	19.4	78.6	>72
7	0.2678	N_2	22.8	27.1	56.5	42.75
9	0.2155	N_2	18.4	21.5	69.8	(b)
11	0.245	N_2	20.9	25.25	58.3	57.75 at 950 psi
			2-day RSS hydrogen vessel [f]			
6	0.0358	H_2	6.91	8.78	55.3	53.0
7	0.0262	H_2	5.06	6.45	73.5	(b)
8	0.0236	H_2	4.55	5.85	(b)	>72.0
9	0.0350	H_2	6.75	8.60	56.4	56.5
10	0.0216	H_2	4.17	5.35	88.5	(b)
11	0.0260	H_2	5.01	6.38	74.0	>72.0
12	0.0191	H_2	3.68	4.65	107.0	(b)
13	0.0429	H_2	8.26	10.44	46.0	50.5
14	0.0408	H_2	7.88	9.99	48.5	(b)
15	0.0320	H_2	6.16	7.86	(b)	(b)
16	0.0270	H_2	5.20	6.64	(b)	(b)
17	0.0283	H_2	5.45	6.95	(b)	(b)
19	0.0350	H_2	6.74	8.60	(b)	(b)
			14 day RSS hydrogen vessel [g]			
6	0.0325	H_2	6.28	9.50	64.1	(b)
7	0.0275	H_2	5.31	8.08	76.3	>72.0
8	0.0458	H_2	8.85	13.45	45.6	40.5
9	0.0316	H_2	6.1	9.25	66.1	52.5 at 300 psi
10	0.0487	H_2	9.4	14.3	43.0	38.0 at 300 psi
11	0.0354	H_2	6.82	10.38	58.8	61.5
12	0.0412	H_2	7.96	12.08	50.8	(b)
13	0.0441	H_2	8.51	12.90	47.7	64.0
14	0.0404	H_2	7.80	11.87	51.7	(b)

[a] All serial numbers are for part number 630094.
[b] Data unavailable.
[c] All serial numbers are for part number 630050.
[d] All serial numbers are for part number 639046.
[e] All serial numbers are for part number 639002.
[f] All serial numbers are for part number 639048.
[g] All serial numbers are for part number 639018.

(4) A requirement for repressurization of the spacecraft after EVA.

High-pressure storage was precluded because of the increased fluid-quantity requirements in conjunction with a need for a minimum-volume and minimum-weight system.

Several new developments and innovations were required for the Gemini CGSS. Among these were the use of aluminized Mylar as a radiant-heat barrier in the vacuum annulus and the use of Fiberglas pads for pressure-vessel supports. Thermal-protection methods had to be improved greatly so that the hydrogen that was required could be stored for the duration of the mission. New manufacturing techniques had to be developed to fabricate this system, which was the first flight-weight supercritical CGSS.

Apollo Command and Service Module Cryogenic Gas Storage System

On July 2, 1960, the House Committee on Science and Astronautics recommended "a manned lunar expedition within this decade." On May 25, 1961, President John F. Kennedy made a national goal of the lunar landing and a safe return of the astronauts before the end of the decade. The primary objectives of the Apollo Program are given in the following list.

(1) Land two men on the Moon and return them safely to the Earth

(2) Perform selenographic survey and sampling

(3) Deploy and use the Apollo Lunar Scientific Experiment Package (ALSEP); this package contains equipment such as the magnetometer, seismometer, solar-wind detector, and so forth

The electrical power system fuel cells and the ECS are necessary for the accomplishment of the goals just described. Just as is the Gemini spacecraft, the Apollo spacecraft is equipped with supercritical cryogenic gas storage systems for oxygen and hydrogen. The oxygen and hydrogen are used for the EPS fuel cells and ECS in the same manner as was discussed previously for the Gemini spacecraft.

The Apollo CGSS consists of two oxygen and two hydrogen tanks which are designed to meet the requirements of the ECS and the EPS fuel cells. Each storage tank consists of two concentric shells, and the annular space between the shells is evacuated. The oxygen tanks have a load-bearing insulation that consists of alternate layers of foil-Fiberglas and Dexiglas paper. The load-bearing insulation supports the pressure vessel within the outer shell and transmits the loads to the mount-support structure. The hydrogen tanks are equipped with non-load-bearing laminar insulation that consists of gold-plated H-film and a vapor-cooled shield. The pressure vessel is supported by means of a strap mechanism that consists of three equally spaced beam assemblies that are composed of alternate layers of titanium, Fiberglas, and H-film.

The fluid-equilibration-heater system for both the oxygen and hydrogen tanks consists of a perforated cylindrical tube and redundant electric heater elements that are coiled and fastened to the external surface of the tube. An electric motor-fan unit mounted on each end of the tube causes the flow that is necessary for convective heating of the fluid and for maintenance of a homogeneous fluid mixture. A typical Apollo cryogenic oxygen storage tank and Apollo spacecraft gas-storage system are shown in figures 19 and 20. In figures 21 and 22, the hydrogen and oxygen tanks are shown as finished products and as they are installed in the spacecraft. Minimum and maximum

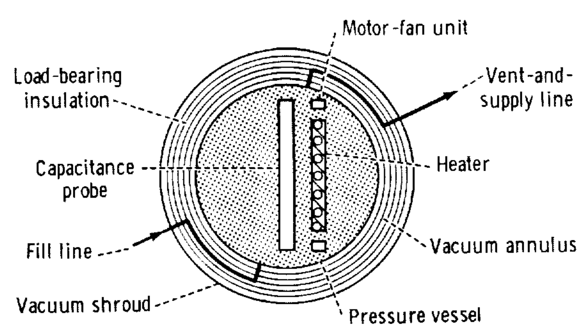


FIGURE 19.—Typical Apollo cryogenic oxygen storage tank.

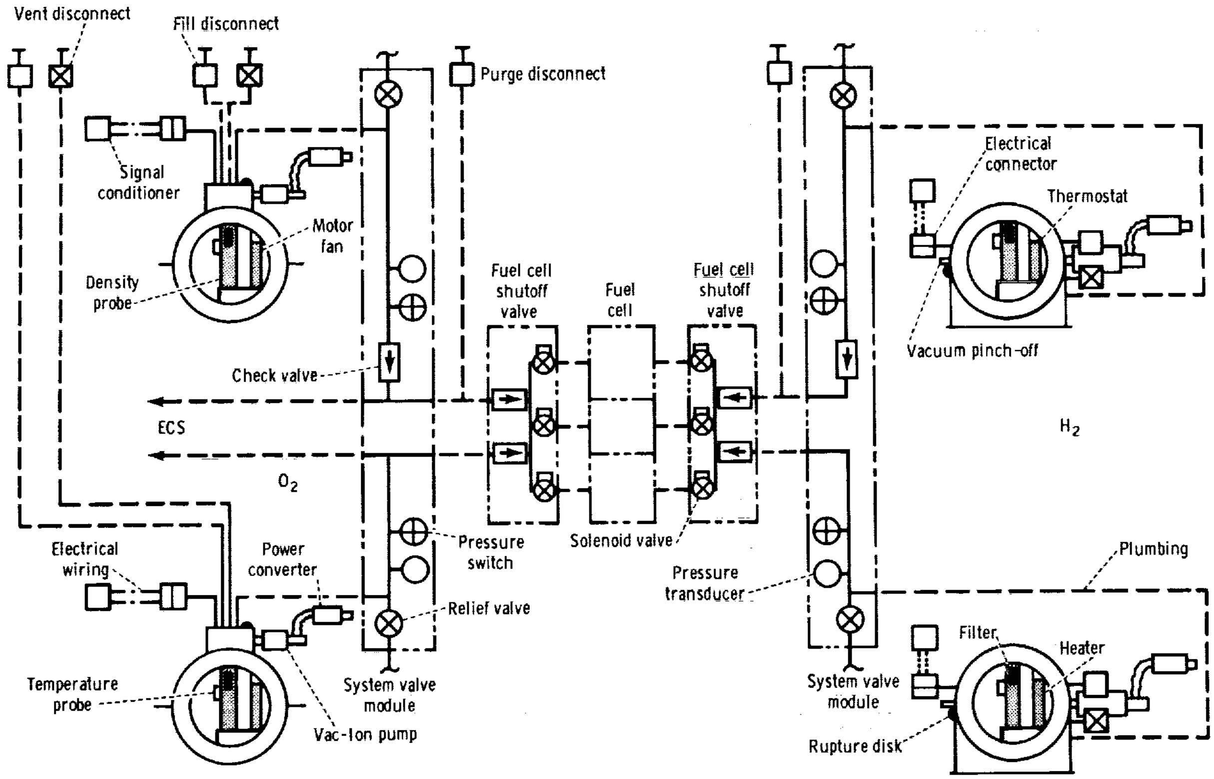
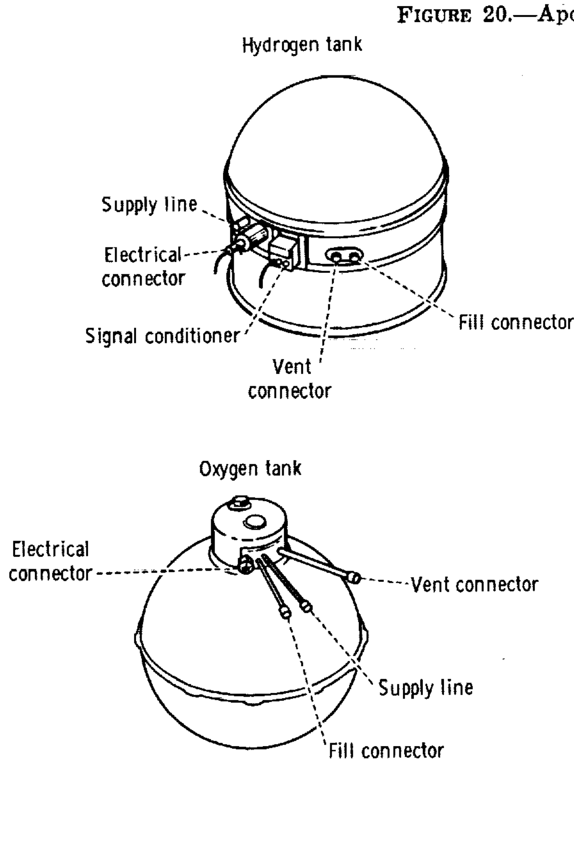


FIGURE 20.—Apollo spacecraft CGSS.



flow rates for hydrogen and oxygen are shown in figures 23 and 24. The minimum flow rate is an important design criteria for any CGSS because it is a measure of how much fluid must flow out of the system so that a constant operational pressure can be maintained. If, at any point in the mission, the demand flow rate (the flow rate required for the ECS and EPS) is less than the dewar minimum flow rate at that same point in the mission, the gas-quantity difference is vented overboard. Because of the cost of placing each pound into orbit, the amount of vented fluid must be minimized. The maximum-flow-rate curves (fig. 24) are illustrative of the maximum amount of fluid that can be expelled and still maintain constant storage pressure. The section on fluid thermodynamics (appendix) contains additional information on these types of curves.

← FIGURE 21.—Apollo cryogenic gas storage tanks.

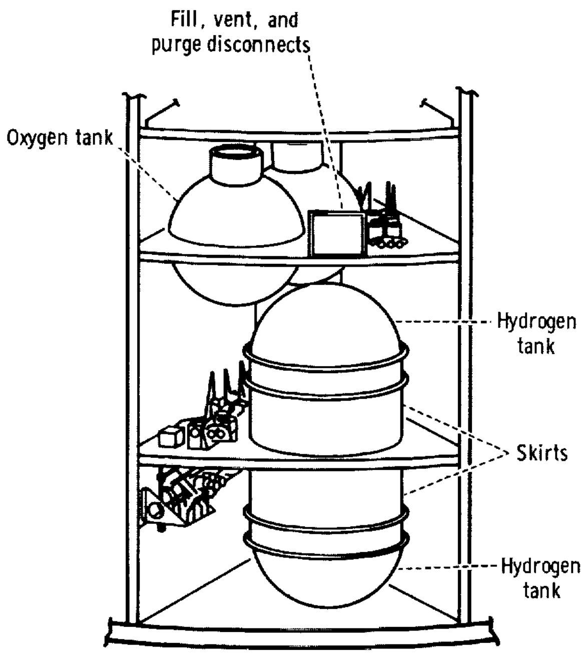


FIGURE 22.—Apollo cryogenic gas storage tanks installed in the spacecraft.

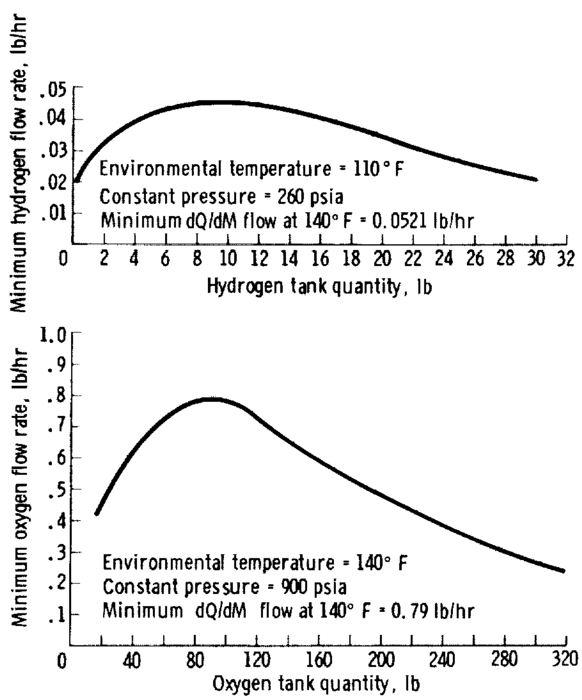


FIGURE 23.—The minimum continuous flow rate at a constant pressure compared with tank fluid quantity for the Apollo CGSS.

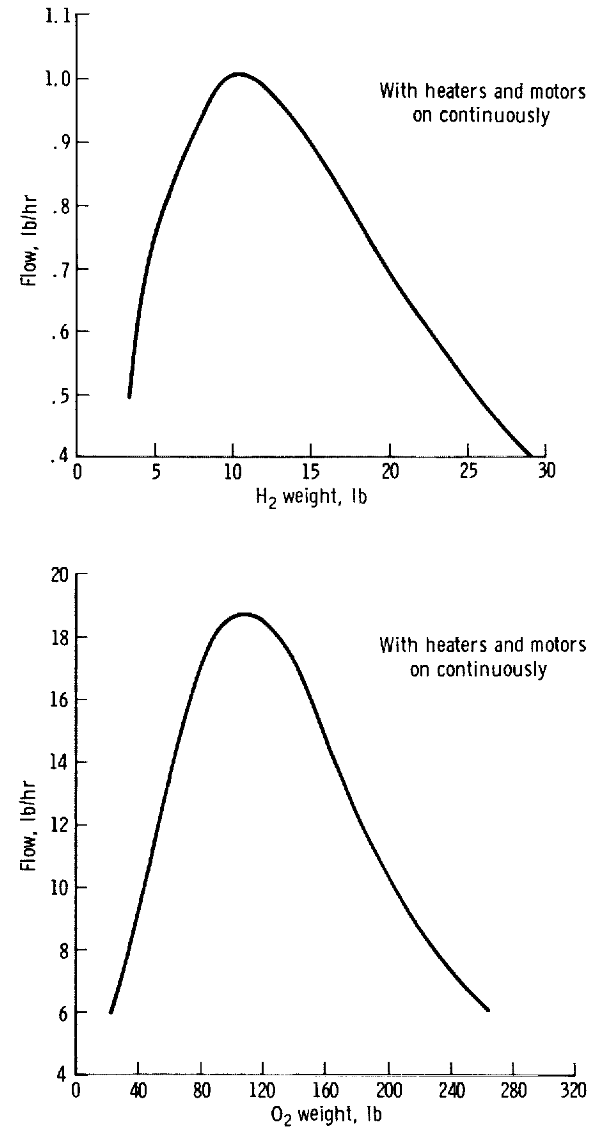


FIGURE 24.—The maximum continuous flow rates for the Apollo CGSS.

In figure 25, oxygen and hydrogen fluid temperatures are plotted against the quantity of fluid that is in the tank. These curves are useful as a cross-check on the accuracy of the capacitance quantity-gaging system during a system test or a flight. The hydrogen minimum and maximum pressurization rates under different ullage conditions are shown in figures 26 and 27. In figures 28 and 29, the oxygen minimum and maximum pressurization rates are shown for different ullage conditions. As used in these figures, ullage refers to the amount that the CGSS, at a pressure of 14.7 psia, lacks being full of liquid. These figures represent the time that will be required to pressurize the system as a function of the ullage. In tables 9 to 11, the CGSS operational, instrumentation, and structural characteristics are given for the oxygen and hydrogen tanks. The component and system weights of the oxygen and hydrogen tanks are itemized in tables 12 and 13. These tables are presented to give anyone who is unfamiliar with the weight of cryogenic gas storage systems an idea of the total weight of a system and the components which make up the major portion of the total weight. The Apollo CGSS was furnished by a contractor under contract to the Apollo spacecraft manufacturer.

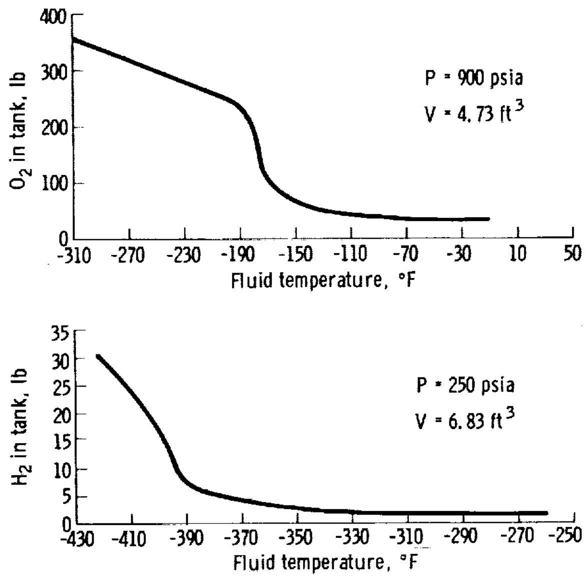


FIGURE 25.—Apollo CGSS tank fluid quantity compared with fluid temperature.

Supercritical storage of the required cryogens was chosen for the Apollo command and service module (CSM) for the same reasons that determined this same choice for the Gemini Program (that is, subcritical storage problems and the large amount of cryogens that were required). It should be noted that much of the Apollo Program development effort was concurrent with the Gemini Program development and produc-

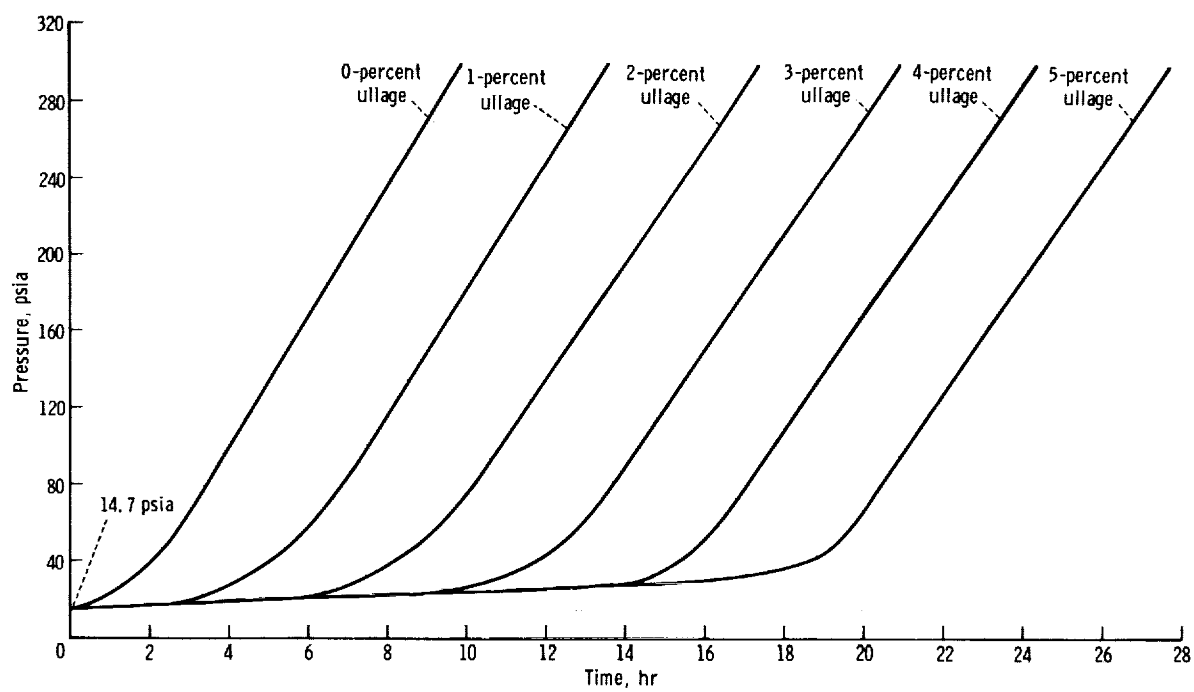


FIGURE 26.—The hydrogen-tank pressurization rate for the Apollo CGSS (no flow, standby, and heaters off).

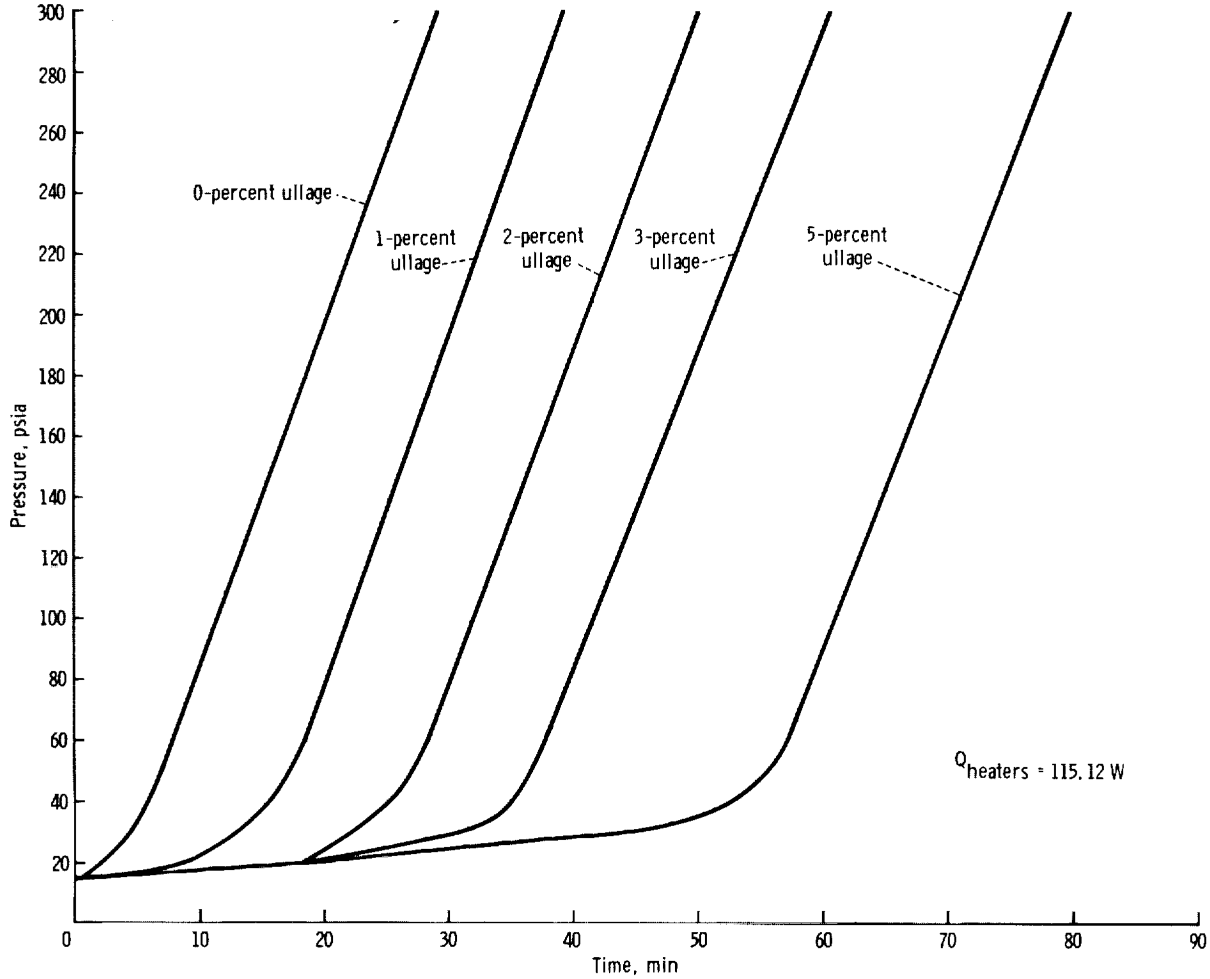


FIGURE 27.—The hydrogen-tank pressurization rate for the Apollo CGSS (no flow and with heaters and motors operating on GSE power).

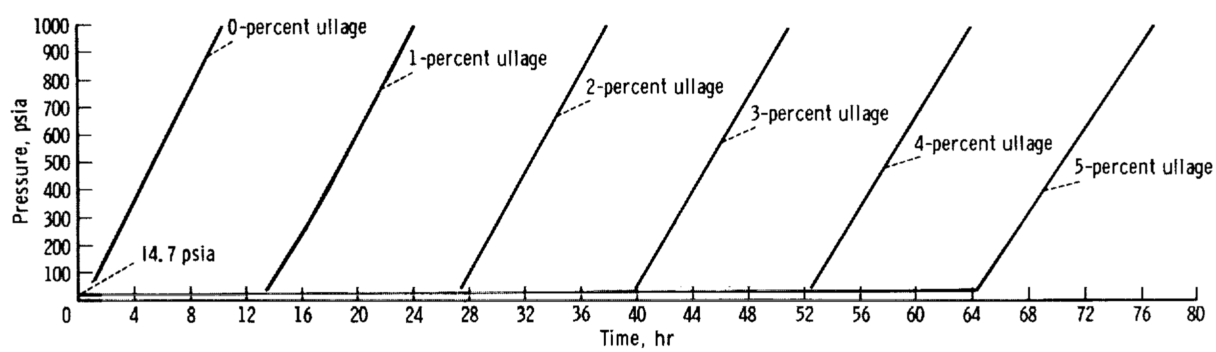


FIGURE 28.—The Apollo CGSS oxygen-tank pressurization rate (no flow, standby, and heaters off).

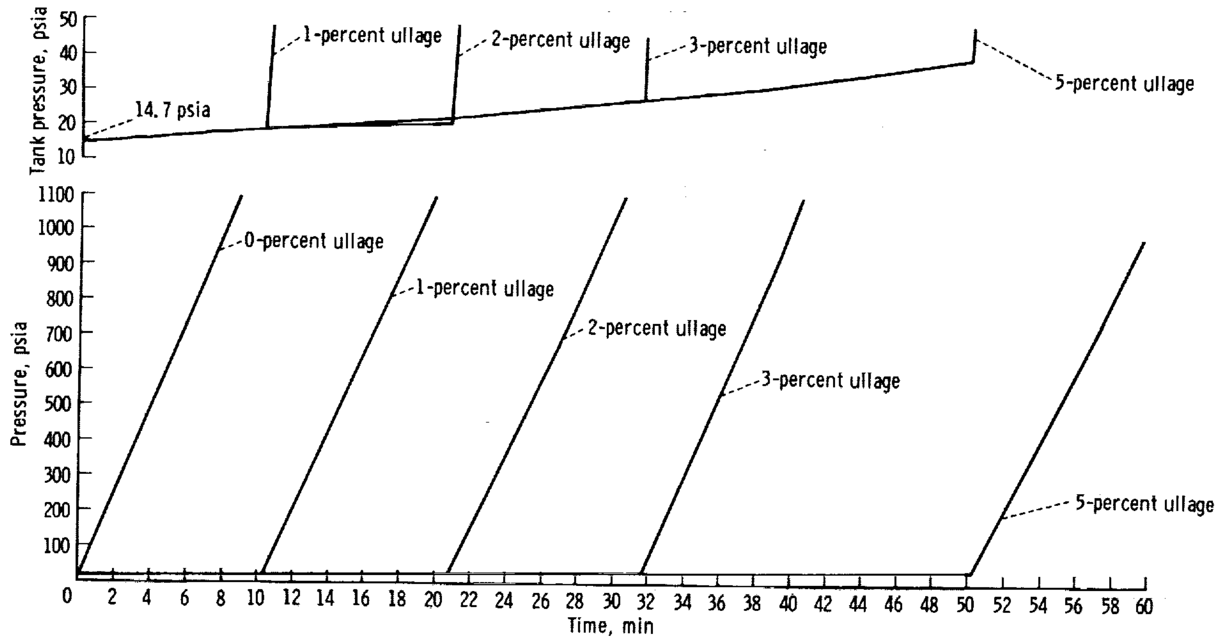


FIGURE 29.—The Apollo CGSS oxygen-tank pressurization rate (no flow and with heaters and motors operating on GSE power).

TABLE 9.—*Apollo CGSS Operational Characteristics*

Characteristic	Hydrogen tank	Oxygen tank
Weight per tank		
Empty (approximate), lb	80.0	91.0
Usable fluid, lb	28.14	323.45
Stored fluid (100-percent indication), lb	29.32	330.1
Ullage (100-percent full indication), percent	4	2
Maximum fill quantity, lb	30.2	336.7
Volume per tank, ft[3]	6.83	4.73
Flow rate per tank		
Specification minimum dQ/dM, lb/hr	0.0725	0.79
Specification maximum, lb/hr	0.135	1.28
Relief-valve maximum flow rate at 130° F environment, lb/hr	6	26
Pressure		
Normal operating, psia	245±15	900±35
Specification minimum dead band of pressure switches, psi	10	30
Minimum operating, psia	100	150
Vent valve [a]		
Crack, psig	273	983
Full flow, psig	285	1010
Reseat, psig	268	965
Temperature		
Operating, °F	−425 to 80	−300 to 80
Delivery, °F	35 to 130	35 to 130
Heater thermostat		
Open, °F	80±10	80±10
Close, °F	60±7	60±7
Servicing		
Fill time, hr	1	1
Conditioning time, hr	4	4
Standby time, hr	30	30
Heat leak (specification)		
Operating, dQ/dM at 140° F, Btu/hr	7.25	27.7

TABLE 9.—*Apollo CGSS Operational Characteristics—Concluded*

Characteristic	Hydrogen tank	Oxygen tank
Standby, Btu/hr	21.00	39.5
Valve-module leakage rate		
External, scc gas/hr/valve	400 (0.736×10^{-6} lb H_2/hr/valve)	400 (9.2×10^{-6} lb O_2/hr/valve)
Interface line sizes		
Fill connections, in.	¼ (0.015 wall)	⅜ (0.022 wall)
Vent connections, in.	¼ (0.015 wall)	¾ (0.028 wall)
Relief connections, in.	3/16 (0.022 wall)	3/16 (0.022 wall)
Feed connections, in.	¼ (0.015 wall)	¼ (0.022 wall)
Outer-shell burst disk		
Nominal burst pressure, psid	90 $^{+10}_{-20}$	75±7.5
Cryogenic valve module		
Feed connections, in.	¼ (0.015 wall)	¼ (0.022 wall)
Fuel-cell-supply connections, in.	¼ (0.022 wall)	¼ (0.022 wall)
Relief-valve outlet, in.	¼ (0.022 wall)	¼ (0.022 wall)
Fuel-cell module		
Feed connections, 2, in.	¼ (0.022 wall)	¼ (0.022 wall)
Fuel-cell-supply connections, 3, in.	¼ (0.022 wall)	¼ (0.022 wall)

[a] Vent valves are referenced to environmental pressure.

TABLE 10.—*Apollo CGSS Electrical and Instrumentation Characteristics*

Characteristic	Hydrogen tank	Oxygen tank
Beech/North American Rockwell interface	Pigtails	Pigtails
Tank connectors	Hermetically sealed pin receptacle	Hermetically sealed pin receptacle
Heaters, 2 elements/tank		
Flight		
Resistance, ohms/element	78.4 (39.2 ohms/tank)	10.12 (5.06 ohms/tank)
Maximum voltage, Vdc	28	28
Nominal power, W/tank	[a] 20	[a] 155
Current at 28 V, A/tank	0.72	5.5
Operating time, percent	10	10
Ground power		
Maximum voltage, Vdc	65	65
Power, W/tank	[b] 108	[b] 836
Operating time, hr	1	1
Pressure switch		
Maximum open pressure, psia	260	935
Minimum close pressure, psia	230	865
Minimum dead band, psid	10	30
Destratification motor fans, 2 each		
Voltage, 3 phase at 400 Hz, Vac	115/200	115/200
Power, W/motor	[c] 3.6	[c] 26.4
Operating time, percent	10	10
Speed [d]		
Flight operation, wet-cold, rpm	4000	1300
Ground checkout, dry-hot, rpm	6000	4000
Pressure transducer		
Range, psia	0 to 350	50 to 1050
Accuracy, percent of full range	±2.5	±2.5
Readability, psi	1.4	3.9
Output voltage, Vdc	0 to 5	0 to 5

TABLE 10.—*Apollo CGSS Electrical and Instrumentation Characteristics—Concluded*

Characteristic	Hydrogen tank	Oxygen tank
Output impedance, ohms	500	500
Power, W	1.5	1.5
Voltage, Vdc	28	28
Quantity-gaging system		
Range, lb	[e] 0 to 28	[e] 0 to 320
Accuracy, percent of full range	±2.5	±2.5
Readability, lb	±0.1	±1.26
Output voltage, Vdc	0 to 5	0 to 5
Output impedance, ohms	500	500
Power at 115 Vac and 400 Hz, W	2.5	2.5
Vac-Ion pump		
Weight, lb/tank	4.2	4.2
Capacity, liters/sec	1	1
Temperature-gaging system		
Range, °F	−425 to −200	−300 to 80
Accuracy, percent of full range	±2.5	±2.5
Readability, °F	0.9	1.6
Output voltage, Vdc	0 to 5	0 to 5
Output impedance, ohms	5000	5000
Power at 115 Vac and 400 Hz, W	1.25	1.25
Solenoid valves (3)		
Voltage, Vdc	28	28
Current, A	2	2
Total flight power		
28 Vdc		
Solenoid, W for 10 sec	168	168
Instrumentation, W for 336 hr	3.0	3.0
Heaters, W for 33.6 hr	40	310
115 V and 400 Hz		
Instrumentation, W for 336 hr	7.50	7.50
Motors, W for 33.6 hr	14.2	102

[a] Equivalent to 68.2 and 530 Btu/hr, respectively.
[b] Equivalent to 368 and 2850 Btu/hr, respectively.
[c] Equivalent to 24.6 and 180 Btu/hr, respectively.
[d] Two-phase operation reduces speed by one-third.
[e] Equivalent to 0 to 4.31 and 0 to 69.5 lb/ft^3, respectively.

TABLE 11.—*Apollo CGSS Structural Characteristics*

Characteristic	Hydrogen tank	Oxygen tank
Material	5Al–2.5Sn ELI [a] Ti	Inconel 718
Ultimate strength, psi	105 000	180 000
Yield strength, psi	95 000	145 000
Young's modulus, psi	17×10^6	30×10^6
Creep stress, psi	71 200	No creep at 145 000
Safety factors		
Ultimate	1.5	1.5
Yield	1.33	1.33
Creep	1.33	(b)
Design stress level, psi	53 000	110 000
Pressure-vessel diameter, in.	28.24	25.06
Pressure-vessel thickness, nominal, in.	$0.044\ {+0.004 \atop -0.000}$	$0.059\ {+0.004 \atop -0.000}$
Outer-shell diameter, in.	31.74	26.48
Outer-shell thickness, nominal, in.	0.033±0.002	0.020±0.002
Support brackets	Aluminum	Inconel
Proof pressure, psi	400	1357
Burst pressure, psi	450	1530

[a] ELI=extra-low interstitial.
[b] Not available.

TABLE 12.—*Apollo CGSS Oxygen-Tank Component Weights (Two Tanks)*

System component	Number required	Total weight, lb
Inner-tank assembly	2	80.6
Hemisphere, lower	2	39.7
Hemisphere, upper	2	40.3
Fan-heater mounts	2	0.6
Fan-heater assembly	2	7.4
Tube assembly	2	3.2
Motor-fan assembly	4	3.5
Heater element	4	0.6
Thermostat	4	0.1
Upper-shell assembly	2	14.6
Upper shell	2	13.8
Housing ring	2	0.8
Lower-shell assembly	2	20.2
Lower shell	2	13.2
Support ring	2	6.4
Seal assembly	2	0.6
Insulation	2	22.9
Fiberglas and Dexiglas paper	2	14.9
Aluminum shields, tubing, and couplings	2	6.6
Miscellaneous	2	1.4
Coil-housing assembly	2	14.4
Housing cylinder	2	1.5
Housing	2	1.6
Diaphragm	2	0.1
Seal assembly	2	0.6
Vac-Ion pump and converter	2	10.0
Insulation	2	0.6
Probe assembly	2	5.14
Tube assembly	2	2.7
Electrical lead adapter	2	0.4
Tube	2	0.01
Adapter	2	0.01
Adapter	2	0.02
Density sensor	2	1.9
Filter	2	0.1
Placard	2	0.008
Adapter electrical connector	2	0.4
Heat shield	2	1.5
Electrical connector	2	0.5
Electrical lead installation	2	0.6
Tank assembly (all preceding components)	2	168.2
Disconnect valves		4.1
Fill valve	3	1.7
Vent valve	2	2.4
Harness assembly—signal conditioner and electrical plug	2	9.34
Signal conditioner	2	3.0
Electrical connector plug	2	0.4
Wire bundle and potting	(a)	5.94
Total system	(a)	181.64

[a] Not applicable.

tion effort. Several developments and innovations that were not available for the Gemini CGSS were available for the Apollo CGSS. Among these developments were destratification fans, vapor-cooled shields, gold-plated H-film, and Vac-Ion pumps. The last section of this chapter contains a functional and conceptual description of each of these items.

Post-Apollo 13 Redesign

During the Apollo 13 mission, a failure occurred in the service module cryogenic storage system. A review board was appointed to investigate the failure, determine the cause or most probable cause, and recommend corrective action to preclude similar occurrences in the future. The review board reported that a short circuit ignited electrical wiring insulation in the number 2 oxygen tank, causing failure of the tank, loss of oxygen in both tanks, loss of electrical power, and abort of the Apollo 13 lunar-landing mission. Also, it was determined that before the Apollo 14 mission, the service module CGSS oxygen tanks should be modified. The ground rules for this modification were as follows.

(1) A third oxygen tank was to be added to the service module to avoid operations in the low-density range.

(2) The use of Teflon, aluminum, and other materials that are potentially combustible in the presence of high-pressure

TABLE 13.—*Apollo CGSS Hydrogen-Tank Component Weights (Two Tanks)*

System component	Number required	Total weight, lb
Pressure-vessel assembly	2	40.35
Tank hemisphere, lower	2	20.00
Tank hemisphere, upper	2	20.00
Heater supports and hardware assemblies	(a)	0.35
Outer shells	4	29.36
Ring installation	2	27.64
Girth ring	2	19.00
Disconnect, fill	2	1.30
Disconnect, vent	2	1.60
Diaphragm, burst	2	0.20
Electrical connector receptacle	2	0.50
Heat shield	2	1.80
Bracketry, hardware, and tubes	(a)	3.24
Vac-Ion pump, magnet, and bracket	2	8.8
Converter	2	1.6
Probe and heater installation	2	12.70
Probe assembly	2	5.30
Density probe	2	3.08
Filter	2	0.20
Wire bundle	2	0.70
Tubes and couplings	(a)	1.32
Heater assembly	2	7.40
Tube and nozzle assembly	2	3.00
Motor-fan assembly	4	3.88
Heater element	4	0.10
Thermostat	4	0.10
Wire bundle	(a)	0.32
Harness assembly	2	6.68
Electrical plug	2	0.58
Signal conditioner	2	3.30
Wire bundle	(a)	2.80
Insulation	2	22.41
Beam assembly	2	5.40
Shield assembly	2	11.48
Spider assembly	4	5.21
Fiberglas, insulation, and clamps	(a)	0.32
Hardware, weld wire, and clamps	(a)	0.70
Tank assembly (all preceding components)	2	150.24
Purge disconnect valve	1	0.60
Support skirt	2	9.15
Total system	(a)	159.99

[a] Not applicable

oxygen was to be minimized throughout the high-pressure oxygen system, and these materials were to be kept away from possible ignition sources.

(3) All electrical wires were to be stainless-steel sheathed and the quantity probe was to be stainless steel instead of aluminum.

The subsequent redesign and analysis effort resulted in a CGSS oxygen-tank design that incorporated each of these items. Additional changes included the incorporation of three heater elements in each tank in place of the original two heater elements, removal of the internal motor fans, and changing the internal temperature-sensor range. A comparison of the changes just discussed with the original Block II Apollo design may be made by reference to figure 19 and table 10.

APOLLO LUNAR MODULE SUPERCRITICAL HELIUM CRYOGENIC GAS STORAGE SYSTEM AND RELATED GROUND SUPPORT EQUIPMENT

The Apollo lunar module (LM) contains a supercritical cryogenic-helium-storage tank used to pressurize the propellant and oxidizer for the LM descent propulsion system. Because cryogenic helium has an extremely low heat of vaporization and has a boiling temperature that is lower than the boiling temperatures of other cryogens that are used for spacecraft applications, helium is much more difficult to transfer and maintain as a liquid. Therefore, the LM helium-storage system requires unique and complex ground support equipment (GSE) for servicing and filling. Because the GSE is unique to the LM helium-storage tank, both the GSE and spacecraft systems will be described. Basically, the GSE and spacecraft units consist of the following major systems:

(1) Spacecraft LM helium-storage tank

(2) The GSE helium-storage and transfer dewar

(3) The GSE helium-conditioning unit

Lunar Module Helium-Storage Tank

The LM cryogenic helium-storage tank is of the typical dewar design. The tank consists of a pressure vessel that has a concentric outer shell; the annulus between the pressure vessel and outer shell is filled with aluminized Mylar insulation and is evacuated to minimize ambient heat transfer into the tank. The pressure vessel is supported within the outer shell by means of Fiberglas pads which transmit the loads to the mount structure. The LM helium-storage tank (fig. 30) includes vacuum-jacketed fill-and-vent couplings, a pressure transducer, and a double burst-disk assembly. If required, system pressure relief is provided by two burst disks that are in series; a vent valve is located between the disks. The vent valve prevents low-pressure buildup between the burst disks in the event the upstream burst disk leaks slightly. The valve is open at pressures less than 150 psia. The valve closes when the pressure exceeds 150 psia, so that for greater leaks the subsequent pressure buildup eventually will rupture the downstream burst disk. If the burst disks rupture, the helium supply is lost, curtailing operation of the LM descent propulsion engine. The LM helium-storage system includes an internal and an external heat exchanger. The internal helium-to-helium heat exchanger maintains the stored helium at the required expulsion pressure. The external fuel-to-helium heat exchanger is used to increase the temperature of the helium supply fluid.

Heat transfer from the outside to the inside of the cryogenic system causes a gradual increase in pressure (approximately 5 to

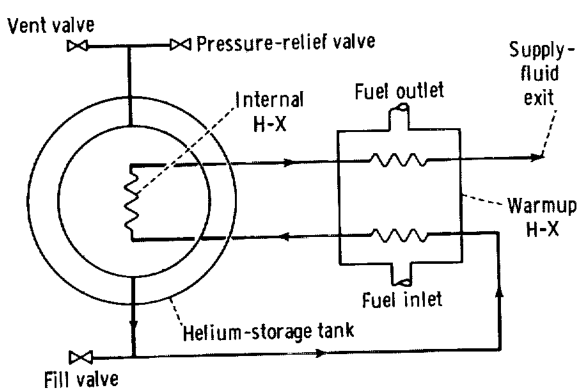


FIGURE 30.—The LM helium-storage system.

10 psi/hr). The initial loading pressure and temperature of the supercritical helium are planned so that the helium will not exceed a predetermined maximum pressure prior to use.

The fluid-flow route of the LM helium-storage system is shown in figure 30. Initially the helium supply fluid passes through the first loop of the external two-pass fuel-to-helium heat exchanger, where it absorbs heat from the fuel. The helium is warmed and routed back through the internal helium-to-helium heat exchanger inside the pressure vessel. The warm helium transfers heat to the remaining supercritical helium in the pressure vessel, causing an increase in fluid pressure; thus continuous expulsion of helium is ensured throughout the period of operation. After the helium passes through the internal helium-to-helium heat exchanger, where it is cooled, it is routed back through the second loop of the fuel-to-helium heat exchanger and is heated before being delivered as the pressurizing agent for the propulsion system fuel and oxidizer tanks. The LM helium-storage system design characteristics are specified in table 14.

TABLE 14.—*Lunar Module Helium-Storage System Design Characteristics*

Characteristic	Value
Maximum fluid fill, lb	48.5
Usable fluid, lb	41.9
Tank-assembly dry weight, maximum, lb	114.0
External fuel-to-helium heat exchanger, maximum, lb	10.6
Tank volume, ft^3	5.95
Tank inside diameter, in.	26.9
Tank outside diameter, in.	33.0
Design operating pressure at 140° R, psia	1710
Proof pressure at 140° R, psia	2274
Burst pressure at 140° R, psia	3420
Burst-disk range, psid	1881 to 1967
Maximum flow rate, lb/min	5.3
Standby time, hr	131
Pressure-vessel material	Ti–5Al–2.5Sn
Outer-shell material	Ti–5Al–2.5Sn

Lunar Module Helium Ground Support Equipment

A helium-conditioning unit and a storage and transfer dewar system are used to service the LM helium-storage tank. The LM helium GSE (fig. 31) is capable of precooling, filling and topping (capacity refill after thermal equilibrium) the LM helium-storage tank. The helium storage and transfer dewar provides cooldown and initial fill of the LM helium-storage tank and also fills the liquid-helium precooler tank in the helium-conditioning unit. The helium-conditioning unit refrigerates ambient helium gas to a near-liquid-helium temperature (8.2° R or lower) prior to delivery to the LM helium-storage tank. Initially the ambient helium gas flows through a coiled, finned-tube heat exchanger that is submerged in liquid nitrogen; thus the gas is cooled to liquid-nitrogen temperature. Then the helium gas passes through the external helium-to-helium regenerative heat exchanger where it is cooled to 16° ± 2° R by the boiloff helium vapors from the liquid-helium precooler. Final cooling is accomplished in the coiled-tube heat exchanger that is submerged in the liquid-helium precooler vessel from which the helium exits at a temperature of 8.2° R or lower. The conditioned helium flows through a fluid-distribution assembly which permits servicing or bypassing of the LM helium-storage tank. All operational modes associated with ser-

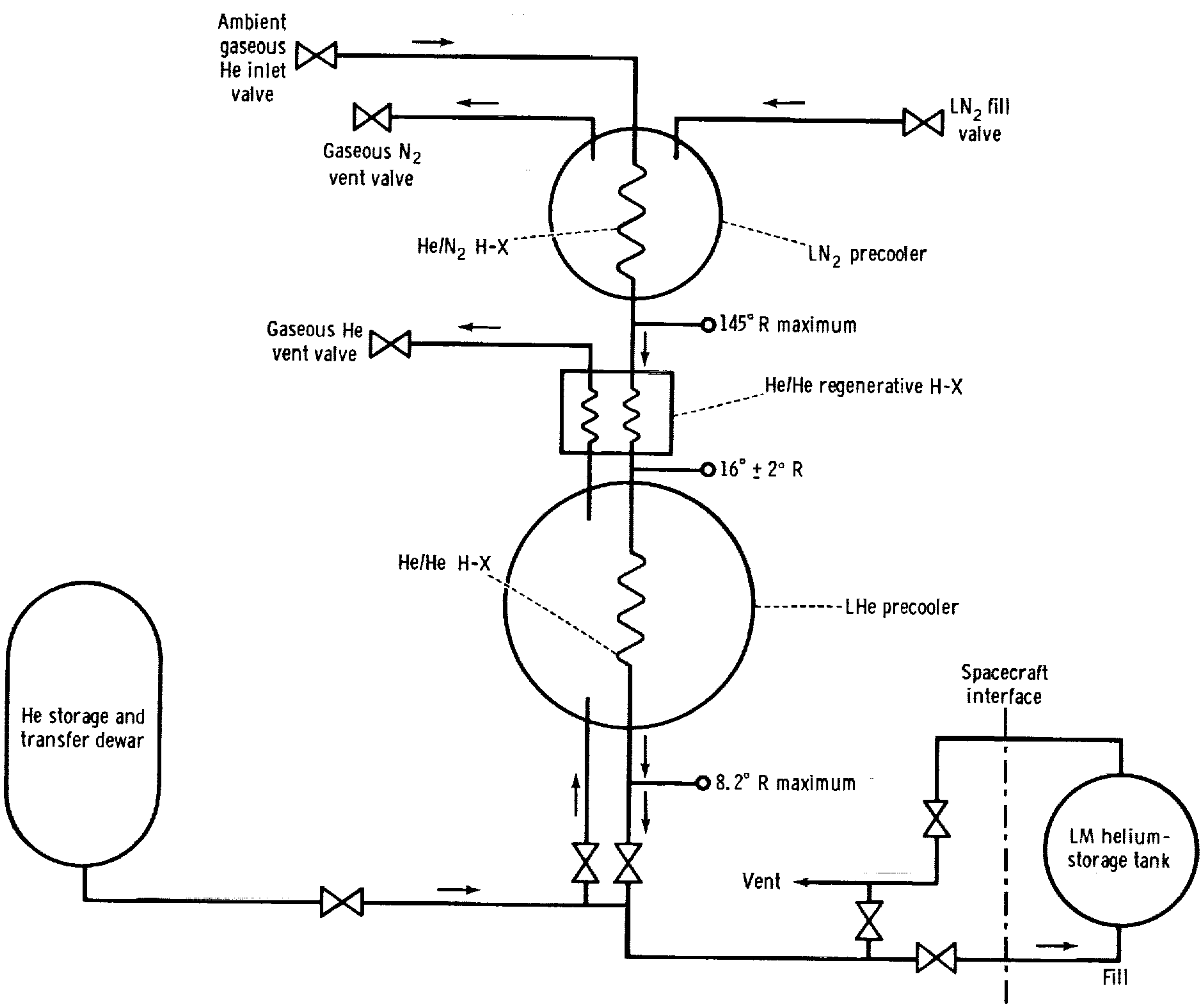


FIGURE 31.—The GSE helium-conditioning and servicing system.

vicing the LM helium-storage tank are controlled and monitored remotely. The nonflight quantity-gaging assembly is used during servicing only; it is attached to the exterior of the storage system by means of quick-disconnect fasteners.

Helium storage and transfer dewar system.—The GSE helium storage and transfer dewar design is similar to spacecraft dewars. The pressure vessel is supported within an outer shell, and the annular space is evacuated to improve thermal protection. A radiation shield is positioned within the vacuum annulus and has provisions for vapor cooling. Boiloff vapor is emitted from the stored helium and is circulated through the small tube that is coiled and fastened to the radiation shield. Radiative heat leak is reduced further by gold-plating the surfaces within the vacuum annulus. The GSE helium storage and transfer dewar system is shown in figure 32; the design characteristics are given in table 15.

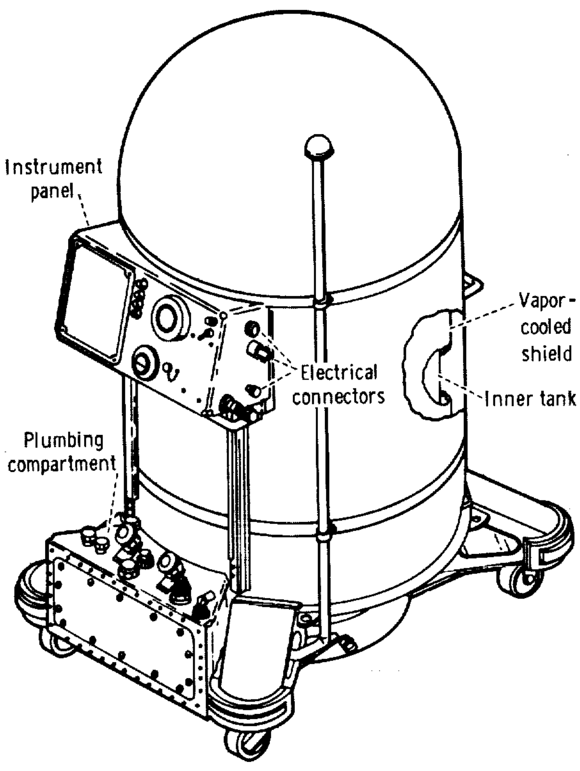


FIGURE 32.—The GSE helium-storage and transfer dewar system.

TABLE 15.—*Design Characteristics for GSE Helium Storage and Transfer Dewar System*

Characteristic	Value
Helium capacity, ft^3	23.6
Vented heat leak at NTP, Btu/hr	1.1
Insulation	Aluminized Mylar Vapor-cooled shield Gold-plated surfaces
System empty weight, lb	355
Envelope size	
Width, in.	36
Depth, in.	50
Height, in.	75
Dewar material	6061 aluminum
Tubing material	Stainless steel

Helium-conditioning-unit system. — The GSE helium-conditioning unit (fig. 33) is a compact and unique system of liquid-nitrogen and liquid-helium containers, heat exchangers, control equipment, and instrumentation. The liquid-nitrogen and liquid-helium containers and the heat exchangers are enclosed within an outer shroud, which makes it possible to evacuate the residual gas so that thermal protection can be achieved. The liquid-helium precooler tank is supported by three equally spaced beam assemblies that are composed of alternate layers of titanium, Fiberglas, and aluminized Mylar. The beam assemblies are connected to the outer shroud through a series of steel cables and Fiberglas compression bumpers. A thin aluminum shield is positioned in the annulus between the pressure vessel and outer shell. This shield, which is cooled by the cold helium gas that is being vented because of heat leak to the pressure vessel, functions as a low-temperature barrier to radiative heat transfer. Additional thermal protection is obtained by the use of aluminized Mylar insulation between the vapor-cooled shield and the outer shroud. The GSE helium-conditioning unit system-design characteristics are specified in table 16. The LM helium-storage tank was fur-

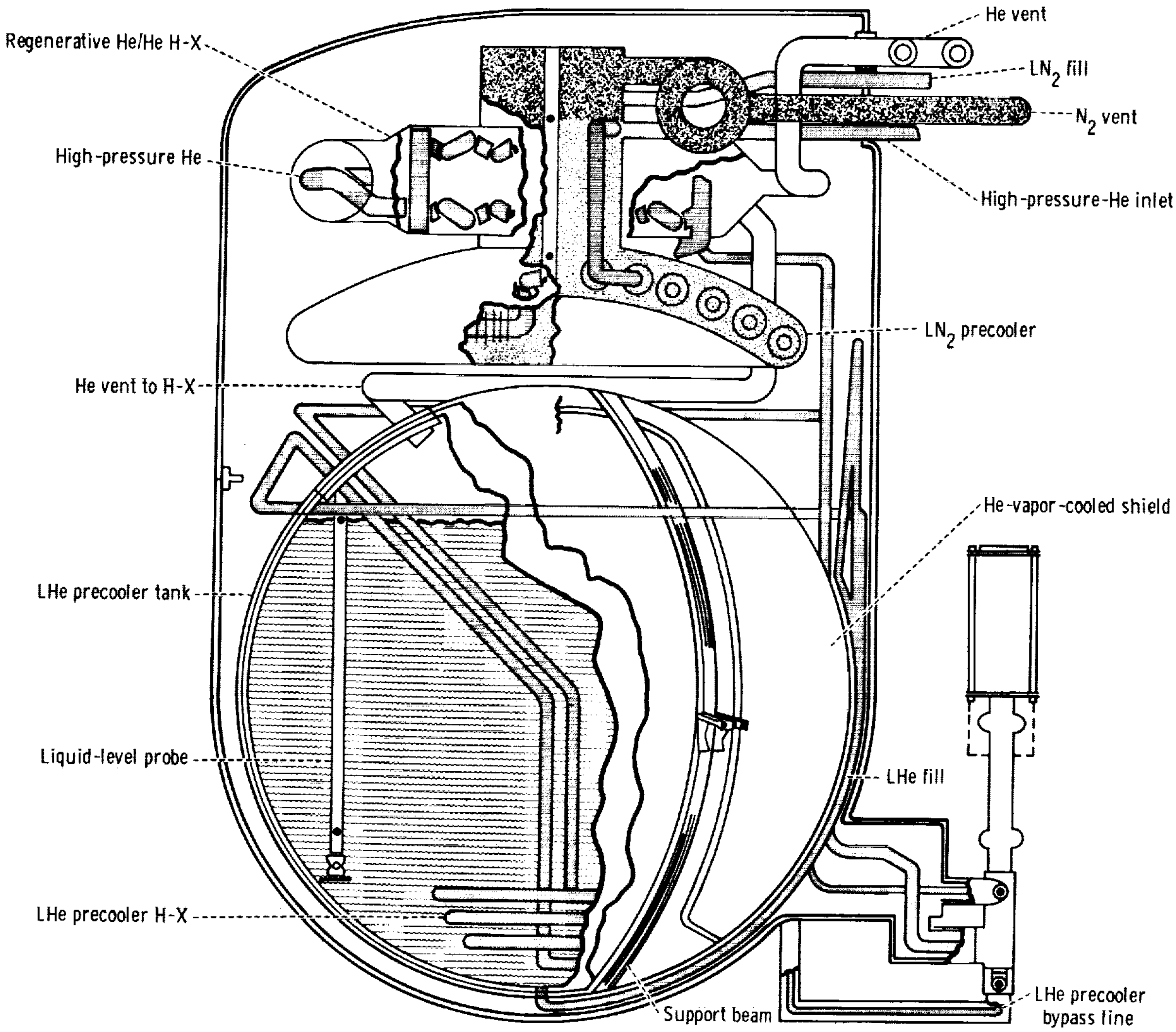


FIGURE 33.—The GSE helium-conditioning unit.

nished by a contractor under contract to the lunar module manufacturer, and the GSE helium storage and transfer dewar and the GSE helium-conditioning unit were furnished by a different contractor under contract to the lunar module manufacturer.

SKYLAB CRYOGENIC GAS STORAGE SYSTEM

After the Apollo Program, the next phase of the manned space flight effort is the Apollo Applications Program (AAP), which has been renamed the Skylab Program. An effort has been made in the Skylab Program to use to the maximum extent possible the hardware and techniques developed in the Apollo Program. One of the basic objectives of the Skylab Program is to establish an environment in which men can live and work under controlled conditions for extended periods of time in space (greater than those associated with Gemini and Apollo). In addition to the Apollo command and service modules, the Skylab will have a workshop (an outfitted Saturn S-IVB launch stage) and a manned solar observatory (known as the Apollo telescope mount, ATM).

The workshop and ATM will be launched together and will arrive in orbit ready for

TABLE 16.—*Ground-Support Equipment Helium-Conditioning Unit System Design Characteristics*

Characteristic	Value
Delivery flow rate, lb/hr	1.25 to 3.25
Delivery-pressure range, psig	35 to 500
Delivery temperature, maximum, °R	8.2
LHe precooler tank	
Sphere diameter, in.	40
Vented heat leak during standby, Btu/hr	3.0
Insulation	Vapor-cooled shield Aluminized Mylar Gold-plated surfaces
Temperature sensors	
For 50° to 180° R	Platinum
For 7° to 60° R	Germanium
Basic material	6061 aluminum
Tubing material	Stainless steel

immediate occupancy and use by the astronauts (who will be launched in a separate vehicle). The ATM will facilitate astronomical observations under conditions that are free from the optical interference of the Earth atmosphere and will function as a platform to demonstrate the ability of man to perform scientific experiments in space. The modified S–IVB workshop will be equipped on the ground prior to launch and will be the living and working quarters for the astronauts while in space. Experiments that are being considered for the workshop include the scientific, technological, applications operation, and medical categories. The experiments will involve the study of human physiological and psychological responses in the space environment and will result in more detailed information on human capabilities for extended manned flights.

Greater cryogen-quantity requirements and mission durations (as many as 56 days) have necessitated the development of larger cryogenic gas storage systems that have better thermal-performance characteristics than previous systems. The Apollo and Gemini cryogenic gas storage systems were not capable of fulfilling the Skylab Program requirements; therefore, a new CGSS design was selected and development began. However, a decision was made to reconfigure the Skylab; this reconfiguration resulted in a decision to terminate the contracts for the AAP CGSS and fuel cell system. The reconfigured AAP will involve the Apollo CGSS and EPS on board the Apollo CSM. High-pressure gas storage will be used for the workshop oxygen and nitrogen supply. Solar cells will be the power source for the workshop.

Because the AAP CGSS design is different and more advanced than the Gemini and Apollo systems, it will be discussed even though the effort has been terminated. Initially the AAP spacecraft was designed to include cryogenic storage dewars for the storage and expulsion of supercritical oxygen, nitrogen, and hydrogen. The oxygen and hydrogen were for the EPS fuel cells, and the oxygen and nitrogen were for the ECS. One of the original design criteria was that the AAP oxygen, nitrogen, and hydrogen cryogenic gas storage systems would be common whenever possible. The reasoning behind this effort to design in commonality was to reduce the number of spares that would be required to support the AAP. The need for spares for all components and systems exists as a major cost factor and time factor in the space program. If the number of spares and unique components can be minimized, then the costs incurred are reduced accordingly.

The oxygen and nitrogen cryogenic gas storage systems were identical in all respects except for the fluid-quantity signal conditioners, which were tailored to the specific cryogen density. The hydrogen CGSS was identical to the oxygen and nitrogen CGSS except for valves, signal conditioner, mount structure, and the number of heaters required. A typical spacecraft system (referred to as a shipset) consisted of three oxygen tanks, three hydrogen tanks, and one nitrogen tank. The oxygen requirements for the ECS and the EPS fuel cells were furnished from the three oxygen tanks; the three hydrogen tanks furnished the required fuel reactant to the EPS fuel cells. Diluent gas for the ECS was supplied by the nitrogen tank.

Each storage tank consisted of two concentric shells, and the annular space between the shells was evacuated. There were two concentric, discrete, aluminum shields that acted as thermal-radiation barriers within the vacuum annulus. The innermost shield on all three types of tanks (oxygen, nitrogen, and hydrogen) has provisions for vapor cooling. The supply fluid passed through a tube that was brazed to this shield prior to exiting the dewar system, thus cooling the shield. This cooling made the shield more efficient in the interception of incoming heat. Part of the intercepted heat was absorbed by the exiting fluid and was carried out of the system.

The pressure vessel was supported by 16 radial bumpers: eight bumpers on the bottom hemisphere and eight bumpers on the top hemisphere. These bumpers were made of a low-thermal-conductivity material known as Kel-F. The pressure-vessel loads were transmitted through the bumpers to the mount structure. The fluid-equilibration-heater system consisted of a perforated cylindrical tube that had coiled electric heater elements fastened to the external surface of the tube. An electric motor-fan unit was mounted on each end of the tube; the unit was a source of convective heating of the fluid, and the unit maintained a homogeneous fluid mixture. The AAP fluid-equilibration-heater system was packaged with the quantity-measuring sensor within one cylindrical tube assembly, whereas the Apollo tanks involved two separate tube structures.

The initial CGSS design necessitated that all three tank types operate at the nominal supercritical pressure of 900 psi, thus making use of common pressure vessels and relief valves. However, during the AAP CGSS contract, it was proved that Inconel 718, the pressure-vessel material, is susceptible to hydrogen embrittlement. A series of tests was conducted at the Manned Spacecraft Center (MSC) in support of the AAP CGSS effort. The results of these tests confirmed the findings of the contractor. However, through the MSC tests it was ascertained that there was a threshold limit to the hydrogen-embrittlement phenomenon. It was determined that Inconel 718 could be used for the hydrogen pressure vessel if the maximum pressure is less than 440 psi.

A typical AAP cryogenic storage tank and storage system are shown in figures 34 and 35. The typical AAP CGSS is shown as a finished product in figure 36. The CGSS design characteristics are specified in tables 17 to 19 for the oxygen, nitrogen, and hydrogen tanks. The major component and system weights are given in table 20. The AAP CGSS was furnished by a contractor as Government-furnished equipment under the management of the MSC.

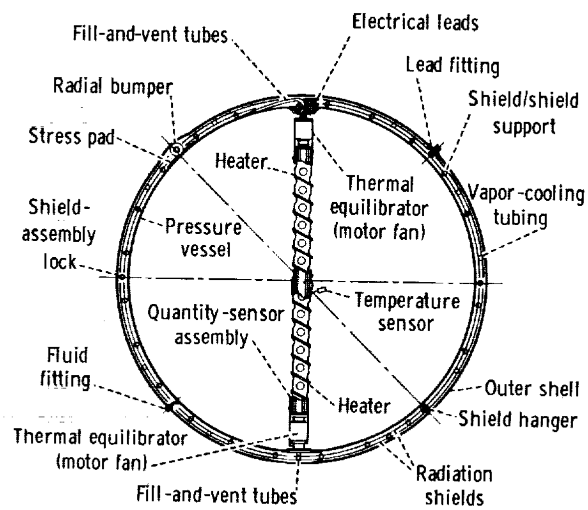


FIGURE 34.—The AAP cryogenic storage tank.

SUPPORTING RESEARCH AND DEVELOPMENT

Most new developments and innovations are begun in research and development programs. The major objective of basic research and development is the generation of new ideas and the development of these ideas so that they can be reduced to practice. Once this has been completed, the ideas can be used in a production contract such as the Apollo Program. Under present modes of management and planning, research and development activity is a continuous process that is quite

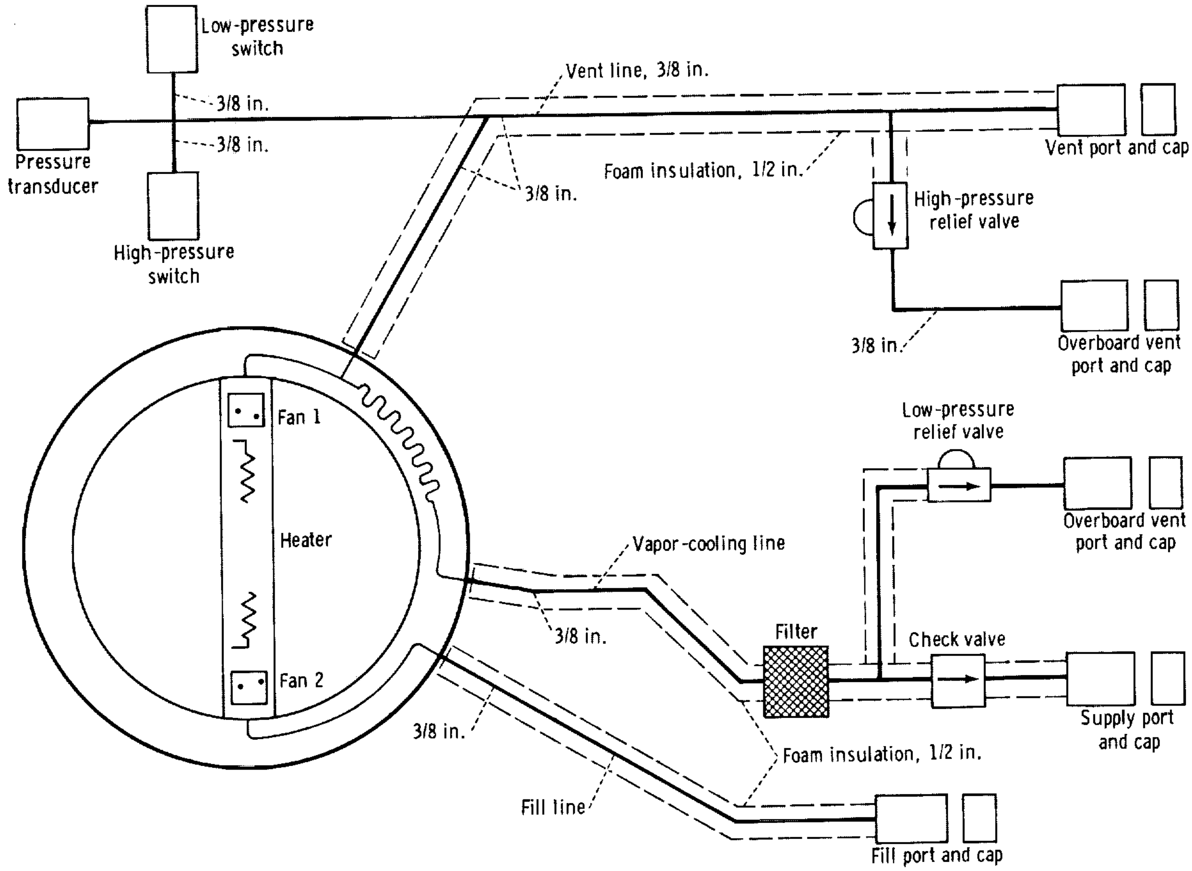


FIGURE 35.—The AAP CGSS.

FIGURE 36.—An external view of the AAP CGSS.

often in direct support of production contracts. Thus, research and development are applied more immediately to and are consciously directed toward the solution of a current problem or toward the fulfillment of an immediate requirement. However, if the technology that is necessary for more advanced CGSS is to be available when it is needed, the research and development process must be initiated and completed long before production contracts are started. The necessary lead time can vary from a few months to many years. The initial development effort by Cailletet and Pictet in the 1870's constitutes one of the earliest supportive research and development efforts for the cryogenic gas storage systems of today. The following sections provide an insight into some of the supporting research and development efforts

TABLE 17.—*Operational Characteristics of the AAP CGSS*

Characteristic	Oxygen tank	Hydrogen tank	Nitrogen tank
Fluid			
Maximum fill, percent	98	98	98
Maximum fill quantity, lb	1221	76.6	868
Usable quantity, lb	1200	75.0	850
Residual quantity, lb	21	1.6	18
Flow rates at normal temperature and pressure			
Minimum normal, lb/hr	0.80	0.06	0.57
Maximum normal, lb/hr	8.0	0.60	8.0
Maximum heat leak at minimum dQ/dM for a 1500-hr mission, Btu/hr	28	5	25
Minimum dQ/dM, Btu/lb	35 at 900 psi	100 at 250 psi	44 at 900 psi
Maximum dQ/dM, Btu/lb	160 at 900 psi	275 at 250 psi	180 at 900 psi
Fluid pressure			
Normal operating range, psia	820 to 910	200 to 260	820 to 910
Minimum delivery, psia	150	100	150
Relief valves [a]			
High pressure			
Crack, minimum, psi	980	430	980
Full flow, maximum, psi	1020	440	1020
Reseat, minimum, psi	950	390	950
Low pressure			
Crack, minimum, psi	950	420	950
Normal flow, maximum, psi	975	430	975
Full flow, maximum, psi	1020	440	1020
Reseat, maximum, psi	920	400	920
Heater circuit			
High pressure			
Open, psia	910^{+0}_{-25}	260^{+0}_{-15}	910^{+0}_{-25}
Close, psia	845^{+25}_{-0}	220^{+15}_{-0}	845^{+25}_{-0}
Low pressure			
Open, psia	845^{+0}_{-25}	240^{+0}_{-15}	845^{+0}_{-25}
Close, psia	820^{+25}_{-0}	200^{+15}_{-0}	820^{+25}_{-0}
Operating fluid temperature, °F	−300 to 80	−425 to 80	−325 to 80
Servicing			
Fill time, hr	3.0	3.0	3.0
Chilldown time, hr	36.0	36.0	36.0
Top-off time, hr	3.0	3.0	3.0
Pressure buildup at normal temperature and pressure			
Standby time, minimum, hr	50	50	50
Heater time, maximum, hr	10	10	10

[a] Pressure above ambient pressure is defined as psi.

that have been conducted regarding cryogenic gas storage systems. Some of the immediate applications of the results of these efforts are presented.

A unique approach to the dewar design involves the use of discrete radiation shields that are suspended within the vacuum annular space in lieu of the multilayer types of insulation. This concept was incorporated into a dewar design, and three prototype

TABLE 18.—*Electrical and Instrumentation Characteristics of the AAP CGSS*

Characteristic	Oxygen tank	Hydrogen tank	Nitrogen tank
Connectors	Hermetically sealed pin receptacle	Hermetically sealed pin receptacle	Hermetically sealed pin receptacle
Heaters			
Voltage, Vdc	28	28	28
Power, each, W	45	45	45
Number	8	1	8
Resistance per heater, nominal, ohms	15	15	15
Power, total, W	360	45	360
Motor fans			
Voltage at 400 Hz, Vac	115	115	115
Power, each, W	25	25	25
Number	2	2	2
Power, total, W	50	50	50
Pressure-gaging system			
Range, psia	0 to 1200	0 to 550	0 to 1200
Accuracy, percent full range	±2.5	±2.5	±2.5
Output voltage, Vdc	0 to 5	0 to 5	0 to 5
Output impedance, ohms	500	500	500
Power, W	0.35	0.35	0.35
Voltage, Vdc	28	28	28
Quantity-gaging system			
Range, percent full	0 to 100	0 to 100	0 to 100
Accuracy, percent of full range	±2.5	±2.5	±2.5
Output voltage, Vdc	0 to 5	0 to 5	0 to 5
Output impedance, ohms	500	500	500
Power, W	4.5	4.5	4.5
Voltage at 400 Hz, Vac	115	115	115
Temperature-gaging system			
Range, °F	−425 to 80	−425 to 80	−425 to 80
Accuracy, percent full range	±2.5	±2.5	±2.5
Output voltage, Vdc	0 to 5	0 to 5	0 to 5
Output impedance, ohms	500	500	500
Power, W	1.1	1.1	1.1
Voltage, Vdc	28	28	28
Ion pump power supply			
Power, W	10	10	10
Voltage, Vdc	28	28	28

systems were developed and tested successfully. The discrete-shield radial-bumper dewar concept was developed and tested by a contractor under the management of the MSC. This concept was used on the AAP CGSS and on the liquid-shrouded CGSS (discussed in a subsequent paragraph).

Discrete shields can be mounted within the vacuum annulus in many different ways; however, only the method that was developed and tested will be discussed. The discrete-shield suspension system involves the pressure-vessel supports and the fluid fill-and-vent lines. The pressure vessel is supported by radial bumpers that are positioned equally on the annular fill-and-vent lines, thus defining and maintaining the annular space. The radial-bumper support concept is illustrated in figure 37. The pressure-vessel loads are transmitted directly through the radial bumpers to the mount structure. A dewar that has discrete shields mounted isothermally within the vacuum annulus is shown in figure 38. The discrete shields are attached

TABLE 19.—*Structural Characteristics of the AAP CGSS*

Item	Oxygen tank	Hydrogen tank	Nitrogen tank
Pressure vessel			
Material	Inconel 718	Inconel 718	Inconel 718
Ultimate strength, psi	180 000	180 000	180 000
Yield strength, psi	145 000	145 000	145 000
Safety factors			
Ultimate strength	2	4.5	2
Yield strength	1.5	3	1.5
Configuration	Spherical	Spherical	Spherical
Volume, ft^3	17.5	17.5	17.5
Outside diameter, in.	39.0	39.0	39.0
Wall thickness,[a] in.	$0.130 {}^{+0.011}_{-0.017}$	$0.130 {}^{+0.011}_{-0.017}$	$0.130 {}^{+0.011}_{-0.017}$
Girth thickness, in.	0.141 ± 0.003	0.141 ± 0.003	0.141 ± 0.003
Weight, lb	182 to 185	182 to 185	182 to 185
Outer shell			
Material	6061 Al	6061 Al	6061 Al
Buckling-pressure differential at 140° F, minimum, psid	20	20	20
Configuration	Spherical	Spherical	Spherical
Outside diameter, in.	41.5	41.5	41.5
Wall thickness, in.	0.064	0.064	0.064
Weight, lb	34.5	34.5	34.5

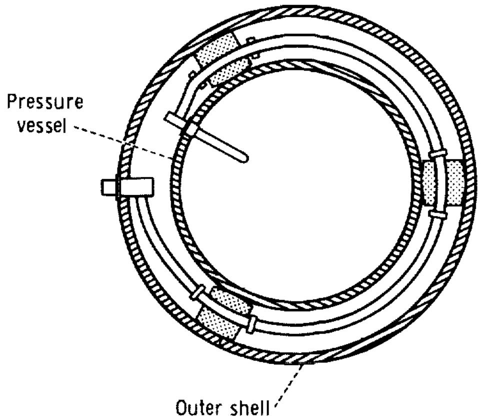
[a] Tolerance varies along the meridian.

TABLE 20.—*Apollo Applications Program CGSS Weights*

Item	Oxygen tank	Hydrogen tank	Nitrogen tank
System weight			
CGSS assembly, lb	380	338	380
Dewar assembly, lb	283	283	283
Mount/interface structure, lb	74	32	74
External components, lb	10	10	10
Interface connections, lb	13	13	13
Major parts weight			
Pressure vessel, lb	182 to 185	182 to 185	182 to 185
Outer shell, lb	34.5	34.5	34.5

to the annular fill-and-vent tubes, and low-thermal-conductivity devices are used for shield attachment and spacing.

Increased vacuum integrity is one of the advantages of a dewar that has discrete shields in lieu of material insulation in the annulus. This increased vacuum integrity is a result of minimum obstruction to molecular migration and removal during system bakeout. The absence of materials such as Mylar (which are affected adversely by heat) facilitates the use of a higher bakeout temperature, thereby making the bakeout process more effective. Thermal protection of the dewar is improved by plating the shields, the outer surface of the pressure vessel, and the inner surface of the outer shell with low-emissivity materials such as silver, copper, or gold. A typical discrete-shield radial-bumper cryogenic gas storage tank is shown in figure



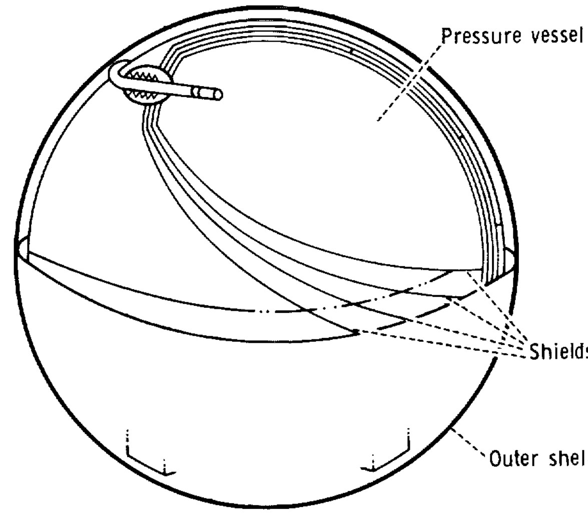


FIGURE 38.—Discrete-shield radial-bumper design schematic.

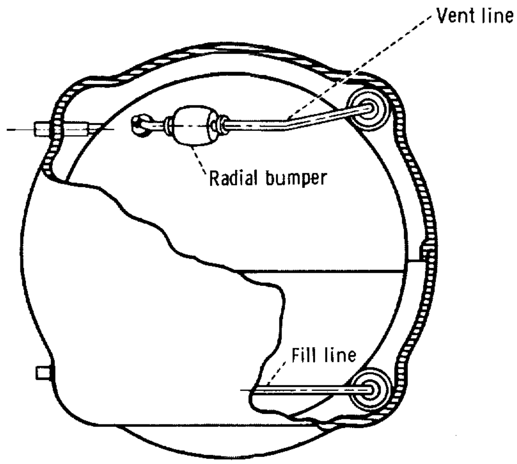


FIGURE 37.—Radial-bumper design schematic.

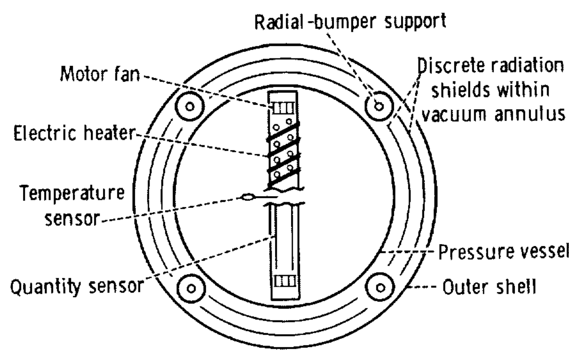


FIGURE 39.—Typical discrete-shield radial-bumper cryogenic gas storage tank.

39. The discrete-shield radial bumper CGSS is shown as a finished product in figure 40. The discrete-shield radial-bumper design characteristics are specified in table 21 for the three cryogenic gas storage systems that were developed and tested.

Experimental Subcritical System

On July 5, 1966, a subcritical nitrogen CGSS was flown as an experiment on the Apollo-Saturn 203 flight. This experiment was the culmination of a program that was directed toward the development of a cryogenic dewar which would deliver warm vapor from two-phase (subcritical) storage in a low-gravity environment, regardless of the liquid orientation. The subcritical mode of storage has the following potential advantages over single-phase (supercritical) storage.

(1) Relief of storage-dewar thermal-design limits because of an increase in the total allowable specific-heat input to the fluid

(2) A substantial weight savings as a direct result of lower operating pressures which facilitate the use of lighter components (for example, in the pressure vessel)

(3) High-density-liquid delivery for refilling portable environmental systems

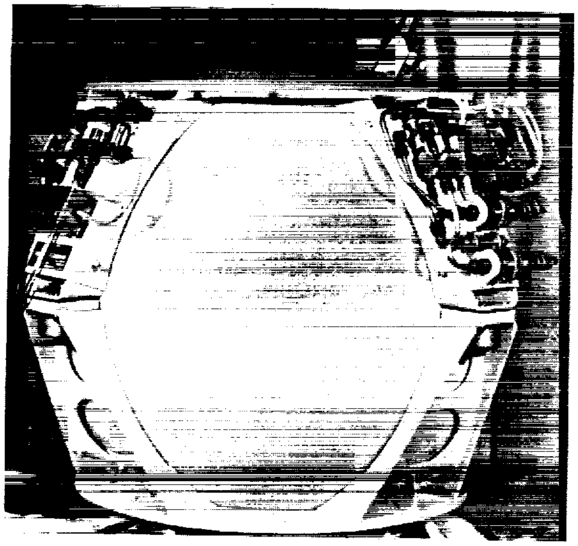


FIGURE 40.—Discrete-shield radial-bumper CGSS.

As was noted, subcritical storage has several advantages over supercritical storage. However, subcritical storage does have two distinct disadvantages: a requirement for specific orientation of the cryogen for purposes of withdrawal or venting and the current lack of an accurate quantity-gaging system. A homogeneous fluid, such as the fluid in a supercritical system, does not have to be oriented specifically, because the fluid occupies the entire pressure-vessel volume. Conventional gaging methods (such as capacitance probes) are inaccurate in a subcritical system in a zero-gravity environment. The experimental subcritical program was directed toward the partial solution and understanding of these problems.

To ensure delivery of vapor only, the delivery circuit had a phase-control heat

TABLE 21.—*Discrete-Shield Radial-Bumper CGSS Design Characteristics*

Characteristic	Oxygen, phase A	Oxygen, phase B	Hydrogen, phase B
Fluid			
Maximum fill quantity, lb	174.5	327	28.5
Percent fill, maximum	98.8	97	97
Heat leak at normal temperature and pressure, vented			
Oxygen test fluid			
Non-vapor-cooling, Btu/hr	7.3	11.7	(a)
Nitrogen test fluid			
Non-vapor-cooling, Btu/hr	6.8	10.3	9.1
Hydrogen test fluid			
Non-vapor-cooling, Btu/hr	5.1	7.9	8.0
Vapor cooling, Btu/hr	(a)	(a)	2.9
Helium test fluid			
Non-vapor-cooling, Btu/hr	(a)	(a)	5.3
Vapor cooling, Btu/hr	(a)	(a)	1.7
Operating pressure range, psi	900	800 to 815	235 to 250
Pressure-relief valve			
Cracking pressure, psi	950	900±10	285±5
Reseat pressure, psi	(a)	810±10	270±5
Heater circuit			
Open, psi	(a)	815±10	250±5
Close, psi	(a)	800±10	235±5
Heater	None	Present	Present
Type	(a)	Electrical resistance	Electrical resistance
Element	(a)	Nichrome	Nichrome
Insulation	(a)	Magnesium oxide	Magnesium oxide
Sheath	(a)	Stainless steel	Stainless steel
Element size, in.	(a)	65 long by 0.062 o.d.	24 long by 0.062 o.d.
Voltage at 60 Hz, V	(a)	115 ac	28 dc

TABLE 21.—*Discrete-Shield Radial-Bumper CGSS Design Characteristics—Concluded*

Characteristic	Oxygen, phase A	Oxygen, phase B	Hydrogen, phase B
Power, each, W	(a)	171.5	30
Number	(a)	2	2
Quality sensor	(a)	Interleaf plate capacitor	Interleaf plate capacitor
Temperature sensor	Thermocouple, copper constantan	Thermocouple, copper constantan	Thermocouple, copper constantan
Motor fan			
Voltage at 900 Hz, Vac	(a)	200	200
Power, each, W	(a)	5.0	5.0
Number	(a)	2	2
Ion pump			
Capacity, liter/sec	0.2	0.2	0.2
Voltage, kVdc	2.9	2.9	2.9
Internal thermal conductor	Spiral 5052 aluminum	(a)	(a)
Pressure vessel			
Volume, ft^3	2.50	4.8	6.7
Inside diameter, in.	20.3	25.1	28.1
Material	Inconel 718	Cryoformed 301 stainless steel	Inconel 718
Wall thickness, in.	0.084	0.036	0.028
Proof pressure, psi	1600	1275	430
Burst pressure, psi	2100	1500	500
Outside surface	Silver plated	Silver plated	Silver plated
Radial bumpers			
Number	6	8	8
Material	Glass-filled Teflon	Kel-F	Kel-F
Fill-and-vent tubing			
Material	304L stainless steel	304L stainless steel	304L stainless steel
Outside diameter, in.	0.3125	0.3125	0.3125
Wall thickness, in.	0.016	0.010	0.010
Radiation shields			
Number	2	2	4
Material	5052 aluminum	6061 aluminum	6061 aluminum
Thickness, in.	0.017	0.020	0.020
Surfaces	Silver plated	Silver plated	Silver plated
Outer shell			
Outside diameter, in.	22.0	28.96	32.36
Material	304L stainless steel	304L stainless steel	304L stainless steel
Wall thickness, in.	0.035	0.033	0.037

[a] Not applicable

exchanger brazed to the pressure vessel. The function of the phase-control heat exchanger was to vaporize any liquid exiting the inner vessel. In a low-gravity environment and in a symmetrically shaped vessel, it is almost impossible to predict whether liquid or vapor will exit the internal regulator valve during two-phase operation. However, because the pressure in the delivery line and in the phase-control heat exchanger is less than the storage pressure, the boiling point of the fluid in the phase-control heat exchanger is lower than the boiling point of the stored fluid. Because of this temperature difference, heat from the stored fluid was transferred to the fluid that passed through the phase-control heat exchanger. Thus liquid that exited the internal regulator valve was evaporated in the phase-control heat exchanger, and left the phase-control heat exchanger as a vapor. Not only was vapor delivery from the phase-control heat exchanger ensured, but heat was removed from the stored fluid.

The overall thermodynamic effect of the phase-control heat exchanger was equivalent to achieving vapor withdrawal from the inner container. During delivery, when the stored fluid was in the compressed liquid state, the same phenomenon occurred, and the exiting liquid was evaporated in the phase-control heat exchanger. Delivery was initiated by the opening of a solenoid valve in the delivery line. When the pressure in the delivery line dropped below normal, the internal regulator valve opened to allow flow into the phase-control heat exchanger. By this means, the delivery-line pressure was increased. The absolute-pressure regulator in the delivery line controlled the pressure of the fluid as it left the system. An electric heater was used to assist in the pressurization of the stored fluid. Instrumentation was used to measure the fluid temperature and pressure at different locations, and to measure fluid-flow rate, heater current, and fluid quantity. The fluid quantity within the pressure vessel was measured by means of a matrix-capacitance gage.

A subcritical CGSS is shown in figure 41. Subcritical CGSS design parameters are shown in table 22. The subcritical nitrogen CGSS was furnished by a contractor under the management of the MSC. The developments, data, and experience that were gained in this effort will have direct application to future space efforts (such as the space shuttle). The advantages that are to be gained from the storage of cryogens in a subcritical condition will be realized whenever the problems are solved. The subcritical pro-

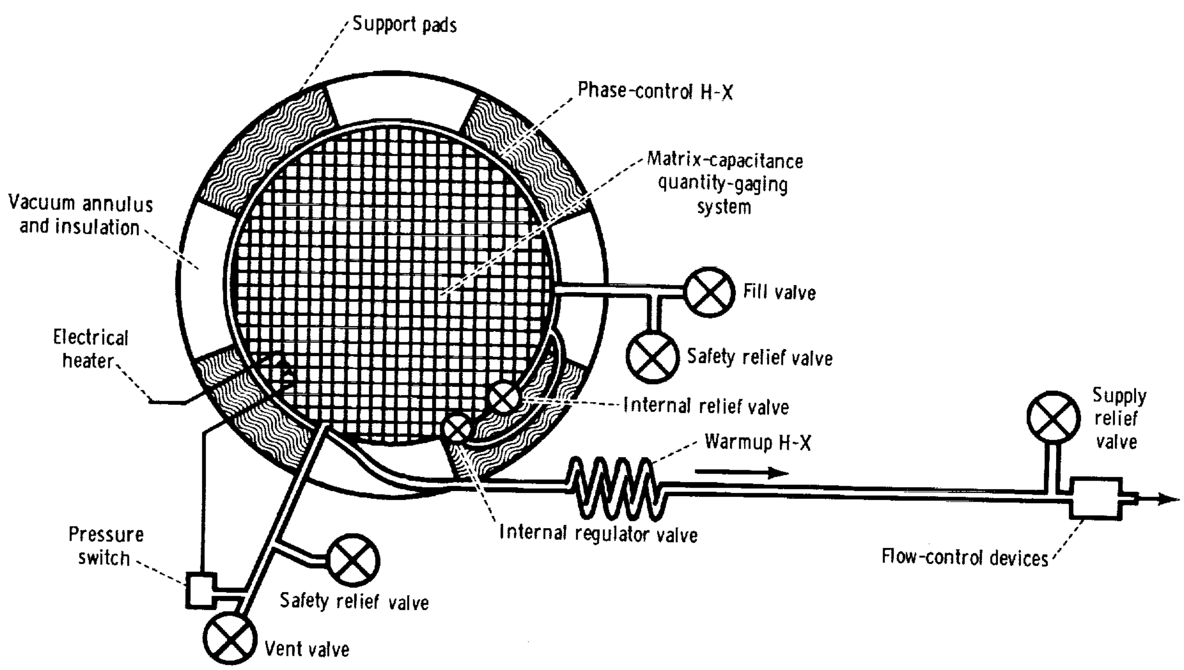


FIGURE 41.—Schematic of the subcritical nitrogen CGSS.

TABLE 22.—*Design Parameters for a Subcritical Nitrogen CGSS*

Parameter	Value
Operating pressure range, psia	130 to 170
Delivery pressure, psia	60±5
Relief pressure, psia	220±20
Proof pressure, minimum, psia	360
Burst pressure, minimum, psia	480
Maximum filled weight, lb	275
Liquid-nitrogen capacity, lb	128
Standby time, nonvented at 160° F, hr	30
Ambient temperature range, °F	−65 to 160
Delivery rate, lb/hr	1.25±0.13
Total 28 Vdc electrical power available, prelaunch, W	400
Total 28 Vdc electrical power available, postlaunch, W	174
Shelf life, yr	1
Radio-noise specification	MIL–I–26600 (USAF)

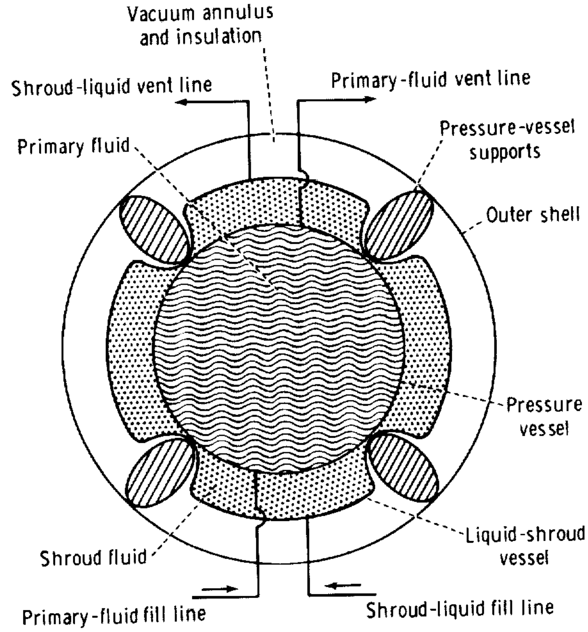


FIGURE 42.—Typical liquid-shrouded cryogenic gas storage tank.

gram just described is one step in the developmental process that is necessary to solve those problems.

Liquid-Shrouded Cryogenic Gas Storage System

A liquid-shrouded CGSS has not been used on any spacecraft to date; however, this concept has been developed and tested. A liquid-shrouded tank is a dewar that has been constructed so that the innermost vessel, containing the primary cryogenic fluid, is surrounded by a concentric vessel which contains a secondary cryogenic fluid. This basic liquid-shroud arrangement is enclosed concentrically within an evacuated annulus which may contain discrete radiation shields or the laminar type of insulation as additional thermal protection. A liquid-shrouded cryogenic gas storage tank is shown in figures 42 and 43. The function of the liquid shroud is to reduce radiative and conductive heat transfer to the cryogen in the innermost tank. To accomplish this goal, the shroud cryogen is sacrificed by allowing it to intercept, absorb, and remove heat that otherwise would have reached the primary cryogen. Heat entering the system must pass through the vacuum annulus, any conventional insulation, and the secondary fluid before it can reach the primary fluid. The shroud fluid is maintained at a saturation temperature that corresponds to a controlled pressure at which the liquid is allowed to evaporate (boil off), hence removing the incoming heat from the system before it reaches the primary fluid. The boil-off vapor may be used further by routing it through a vapor-cooled shield in the vacuum annulus, thus intercepting and removing heat before it reaches the shroud fluid.

The basic premise of the shroud concept is to protect thermally a primary fluid by the placement of a barrier of an expendable secondary fluid between it and the heat source (the environment). A primary and secondary fluid are called a fluid pair. One of the major considerations in the selection of a suitable fluid pair is that any solidification of the stored fluid would result in a severe safety problem. Therefore, a secondary fluid must be chosen that has a higher boiling point than the triple point of the primary

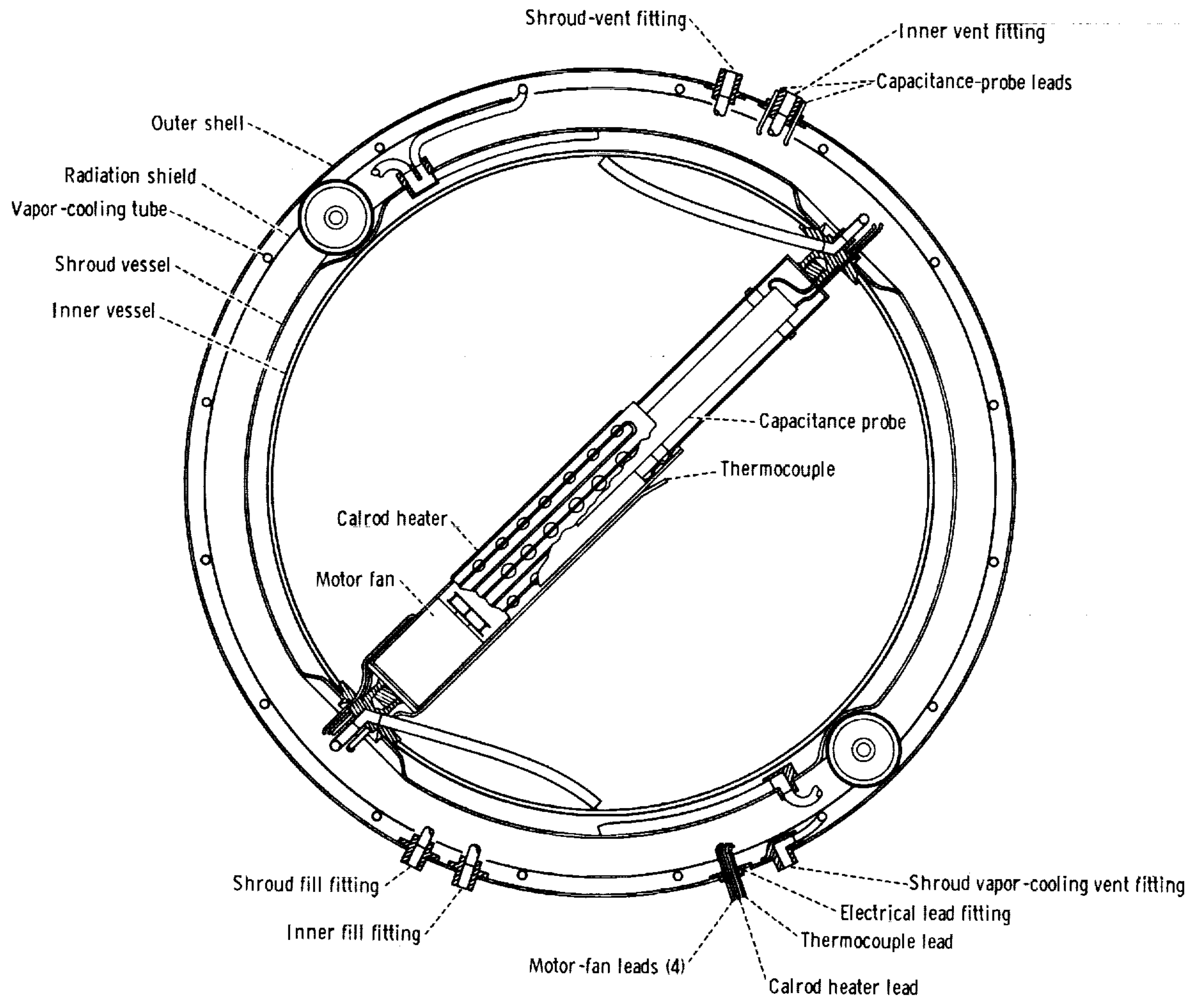


FIGURE 43.—Shroud tank assembly.

fluid. Another hazard that should be avoided is the selection of a fluid pair that can form an explosive mixture (such as oxygen and hydrogen). Although there are no direct connections between the containers that hold the two fluids, there is the ever-present possibility of leakage. Also, the vented vapor of the two fluids can mix outside the system.

The capability of particular shroud fluids to remove heat from the CGSS must be considered. For spacecraft applications, the total system wet weight (which includes fluid weight) is a prime consideration. Some fluids that may involve helium as the primary fluid of the fluid pair are hydrogen, helium, nitrogen, oxygen, neon, and argon. These fluid pairs are listed in order of descending desirability according to their heat-removal capability in conjunction with minimum system wet weight. Helium/hydrogen is the most desirable fluid pair. The liquid-shrouded dewar is useful because it functions as a very effective thermal barrier and it is useful for applications that require extremely low loss rates. Liquid-shrouded systems have the most direct application to storage-duration requirements that are between 500 and 1000 hours, such as a helium-pressurization system for an extended lunar stay. The liquid-shrouded CGSS was developed and tested by a contractor under the management of the MSC.

Single-Wall Tank

The single-wall tank is a CGSS that does not involve the vacuum annulus of a dewar. A single-wall CGSS has a pressure vessel (which contains the cryogen) and an external thermal-protection system. Unlike a dewar system, the pressure vessel has external thermal insulation but does not have a vacuum annulus.

A single-wall tank was fabricated and tested as a research tool (fig. 44). This particular single-wall CGSS consisted of a pressure vessel enclosed within a two-component insulation system. The ground-hold insulation that was designed to protect the system from servicing and prelaunch thermal environments consists of a 2-inch layer of polyurethane foam bonded to the outer wall of the pressure vessel. The orbital insulation that was designed for thermal protection in space environments consists of 100 layers of aluminized Mylar wrapped on the outside of the foam insulation at a density of 50 layers per inch. The tank is supported by Fiberglas tension rods that are attached to an aluminum frame assembly. The single-wall CGSS and its support structure were designed for use as a research system only and were not capable of being used as a flight system without additional design and modification.

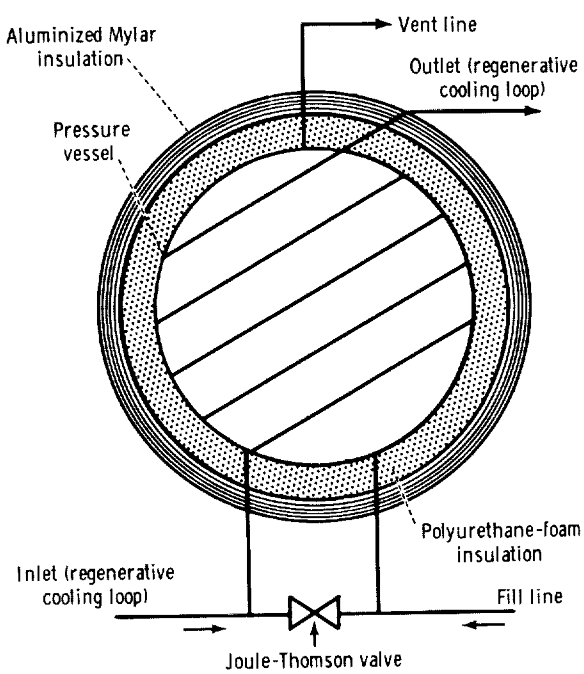


FIGURE 44.—Single-wall-tank CGSS.

The single-wall-tank design included a regenerative cooling system that consisted of a Joule-Thomson expansion valve and an external cooling coil (which served the same purpose as a vapor-cooled shield) that was directly attached to the pressure-vessel external surface. Other components that were located within the pressure vessel included a fluid-quantity sensor, a fluid-temperature sensor, and an electrical heater. Because the insulation is exposed to the atmosphere, condensate may accumulate between the layers of the aluminized Mylar when the tank is filled with a cryogen. Tests have proven that excessive moisture will wash the aluminum off the Mylar, thereby degrading thermal-protection efficiency. Therefore, the aluminized Mylar insulation must be purged constantly with a dry inert gas during filling and ground hold. This can be accomplished by placing a plastic bag around the system and filling the bag with dry inert gas. The dry inert gas replaces the terrestrial atmosphere that surrounds the tank, and the inert gas will not condense when the insulation temperature is lowered. During storage, an empty tank may undergo some condensation in the insulation because of ambient-temperature variations and changes in atmospheric conditions.

The single-wall-tank CGSS design characteristics are shown in table 23. The single-wall-tank system just described was developed by a contractor and was tested by the MSC. The knowledge gained from this effort will be one part of the body of knowledge needed for the assessment of the single-wall-tank concept and external insulation. Cryogenic tankage for use on the space shuttle and the space base are possible future applications of single-wall tanks and external insulation.

Honeycomb Outer Shells

There are many equations for the design of a sphere that is to be subjected to external

TABLE 23.—*Design Characteristics of the Single-Wall-Tank CGSS*

Characteristic	Value
Pressure vessel	
Material	Inconel 718
Outside diameter, in.	39.0
Volume, ft^3	17.36
Liquid-nitrogen capacity, lb	875
Thermodynamic performance (with nitrogen)	
Environmental pressure, torr	1.2×10^{-6}
Environmental temperature, °F	86
Fluid pressure, psia	16.6
Heat leak, Btu/hr	8.52
Flow rate, lb/hr	0.1001
Insulation	
Ground hold	2 in. of Nopcofoam (polyurethane foam)
Orbital	2 in. of 0.25 mil NRC–2 (aluminized Mylar)

pressure. The critical pressures calculated from these equations may vary considerably because of variations and imperfections in the shells. As a result, the outer-shell equations must be developed from applicable test data for the specific shell configuration. As the overall size of the outer shells become larger, the shells become increasingly vulnerable to buckling. This condition must be compensated for by the use of a heavier gage material, which results in a weight increase. To overcome the undesirable characteristics of large monolithic outer shells, a honeycomb-sandwich design has been developed and tested. Test results are indicative that the buckling-pressure-to-weight ratios are larger for honeycomb spheres than for monolithic spheres.

The three basic components in a honeycomb sandwich are the facings, core, and adhesives. The facing material consists of the skins, which are attached to the honeycomb core to make up the composite honeycomb structure. The core is the honeycomb-cell structure which makes up the heart of the honeycomb sandwich. The adhesives are the materials which bond the facings to the core. In the selection of facing materials, it is necessary that the material have excellent bonding characteristics. The selected core material should be designed to fit over cylindrical and hemispherical shapes. Epoxy is considered to be the conventional adhesive for honeycomb-sandwich binding, and it must have minimum-outgassing characteristics.

CONCEPTUAL AND FUNCTIONAL DESCRIPTION OF CRYOGENIC GAS STORAGE SYSTEM COMPONENTS

In previous sections of this chapter, the individual components of cryogenic gas storage systems have been referred to extensively. Little attempt has been made to describe or explain the individual functions of each of these components. However, a conceptual and functional description of the major CGSS components should result in additional insight and useful knowledge. The following discussion contains more detailed information than has been given heretofore.

Pressure Vessels

The pressure vessel is the innermost vessel, and it contains the cryogenic fluid at operating pressure. Most pressure vessels are spherical; however, serious consideration is being given to cylindrical vessels that have hemispherical or elliptical ends. Usually, fabrication of the pressure vessel is accomplished either by drawing, forging and machining, spinning, or hydraulic forming techniques. The pressure-vessel membrane thickness is determined readily by theoretical analysis based on known material properties. However, discontinuities such as openings, bosses, and weld areas must be analyzed, primarily by means of empirical methods that are based on applicable test data.

Depending on the system design, internal components such as the quantity sensor, temperature sensor, electrical heater, and thermal conductor may have to be installed prior to welding the pressure vessel. The electrical leads and fluid lines penetrate the pressure vessel through special fittings that are designed according to the temperature, pressure, and leakage requirements. The selection of materials for use in pressure vessels must involve a consideration of the following characteristics at both the cryogenic and the maximum operating temperatures: fracture toughness, strength-to-weight ratio, fluid compatibility, corrosion resistance, formability, joinabilty, fatigue properties, chemical properties, configuration, permeation, creep properties, embrittlement properties (parent and weld materials), joint efficiency, galvanic corrosion on dissimilar-metal joints, application, availability, developmental problems, and cost.

To date, titanium, Inconel, stainless steel, and aluminum are considered to be the most suitable materials for use in cryogenic pressure vessels. The mechanical properties of several materials are given in table 24. It should be noted that titanium A110–AT (Ti–5Al–2.5Sn) is susceptible to room-temperature creep, and that Inconel 718 is susceptible to stress cracking when exposed to gaseous hydrogen. Test results have proven that Ti–5Al–2.5Sn ELI (extra-low interstitial) has a room-temperature creep-strength level of approximately 71 200 psi. Therefore, appropriate design considerations and safety factors must be applied to the creep-strength level and to the ultimate-strength and yield-strength levels. Also, test results have proven that Inconel 718 is susceptible to environmental-stress cracking when exposed to gaseous hydrogen. The results of an MSC test program were indicative that stress cracking would not occur if the maximum operating pressure was limited to 22 percent of a proof-test pressure that involved liquid nitrogen as the pressurant. The liquid-nitrogen proof-pressure test was performed to screen a maximum flaw size in the material at a pressure that was slightly less than the liquid-nitrogen-temperature yield strength of 204 000 psi. Hence, the pressure-vessel design-stress level corresponds to a safety factor of 4.5, based on the room-temperature ultimate strength of the material.

Outer Shells

The outer shell of a dewar encloses the pressure vessel, pressure-vessel supports, insulation, shields, fluid lines, and electrical leads, and it forms the outer surface for the vacuum annulus of a dewar. Therefore, the outer shell must be sound structurally and

TABLE 24.—*Pressure-Vessel-Material Properties at 70° F*

Pressure-vessel material	Tensile strength, ultimate, F_{TU}, ksi	Tensile strength, yield, F_{TY}, ksi	Elongation, percent	Young's modulus, $E \times 10^{-6}$, psi	Density ρ, lb/in^3	$\frac{F_{TU}}{\rho}$
Titanium alloy C 120–AV (annealed)	136	127	12	15.8	0.160	850
Titanium alloy [a] A 110–AT (annealed)	115	105	16	15.6	0.161	715
Rene 41 (solution treated)	200	160	14	31.6	0.298	672
304 ELC stainless steel (SS)	82	30	62	29	0.29	282
304 SS (40-percent cold reduction)	155	130	8	25	0.20	535
AM 350 SS (SCT 1050)	166	138	15	28	0.29	572
301 SS cryogenic formed	260	220	5	28	0.29	897
Aluminum alloy 6061–T6	45	38	18	10	0.098	459
Inconel 718 [b] (double aged)	192	156	18	31	0.298	645
Aluminum alloy 2219–T62	54	39	11	10	0.098	551
Aluminum alloy-Kaiser 7039–T6	63	55	13	10	0.098	643
Aluminum alloy Alcoa X 7006–T6	63	57	15	10	0.098	643
Beryllium	69	59	5	43	0.066	1045

[a] Titanium A 110–AT (Ti–5Al–2.5Sn) is susceptible to room-temperature creep.
[b] Inconel 718 is susceptible to stress cracking when exposed to gaseous hydrogen.

must be compatible metallurgically with a vacuum environment. Because the inner surface of the outer shell is exposed to a high vacuum (10^{-6} torr), the shell must be designed to withstand collapse pressures from the external atmospheric pressure. Primarily, the outer shells are designed to resist buckling types of failures. Because the outer-shell weight is proportional to the material density and is inversely proportional to the square root of the modulus of elasticity, it follows that low density and high elastic modulus are important factors in the selection of the outer-shell material and that the material tensile strength is secondary.

Heaters and Motor Fans

Electrical heaters, motor fans, and thermal conductors are used to control the cryogen pressure and stratification by adding and diffusing heat to the stored fluid within the pressure vessel. Heat is required for maintaining or increasing the fluid pressure for the purpose of expulsion. Thermal conductors or motor fans are required for the maintenance of a homogeneous fluid.

Thermal stratification.—One characteristic of stored cryogenic fluids is termed thermal stratification. A thermally stratified system is not in thermal equilibrium, and an unstable pressure is established by the vapor pressure of the fluid zone that has the highest temperature. Equilibration, or thorough mixing of a thermally stratified system, results in a new equilibrium storage pressure. Stratification is common to all fluids that are subjected to heat transfer. The density layers (strata) are caused by the temperature differences that must accompany heat transfer. In a 1g environment, these layers will segregate according to weight, but in a low-g or 0g environment, they will remain static unless disturbed by some internal or external force. If a supercritical fluid is stratified and then mixed, a pressure decrease will result. Initially heat energy is distributed heterogeneously within the fluid, and this heated portion, in turn, pressurizes the contents of the vessel. Therefore, the resulting pressure is not an equilibrium condition which is achieved theoretically by the homogeneous distribution of heat energy. If a nonequilibrium condition is forced to become an equilibrium condition because of the redistribution of the total available energy, a pressure decrement is the result.

Cryogen stratification is undesirable for three reasons: (1) possible pressure decay below the critical pressure, causing operation in a two-phase regime; (2) possible overheating of the heater; and (3) inaccurate quantity measurement. Because the accuracy of capacitance-type probes is dependent upon uniform fluid density, it is essential that stratification be minimized if such a device is to be used for the measurement of fluid quantity. In a homogeneous fluid mixture, the capacitance probe will sample a uniform medium, and then will indicate an accurate quantity measurement.

Heat-transfer methods. — The two basic methods used for transferring heat within the cryogenic fluid are classified as being either static or dynamic. The static system depends on thermal-conduction from electric heaters and thermal conductor combinations (such as an electrofilm resistive coating on a large internal heat transfer surface) or by means of electric heaters which conduct heat through a large-surface thermal conductor. The dynamic system depends on forced mixing involving convective heat transfer; this is accomplished by means of an internal heater and fan that are powered by an electric motor or by an external pumping loop that circulates the fluid through a heat exchanger that contains a heater.

A static system consists of a thermal conductor, such as one or more copper spheres positioned internally and concentric within the pressure vessel, that is used for distributing the heat and equalizing the fluid temperature. Usually, the thermal conductors have holes for the purposes of weight reduction and fluid passage. The electric heater unit consists of a nichrome-wire resistance-heater element that is insulated by powdered magnesium oxide contained within a metallic sheath. The heater is coiled and is attached to the surface of the thermal conductor. The major advantage of a static heater system is that there are no moving parts; thus, higher reliability usually is achieved. However, cryogen-stratification considerations become more conspicuous, and more attention must be directed toward the thermal-conductor design.

A dynamic system consists of two electric-motor-driven impellers (motor fans) in conjunction with a heater; simultaneously, this device results in destratification and convective heating of the fluid. The internal electric heater (fig. 45) consists of a heater coiled

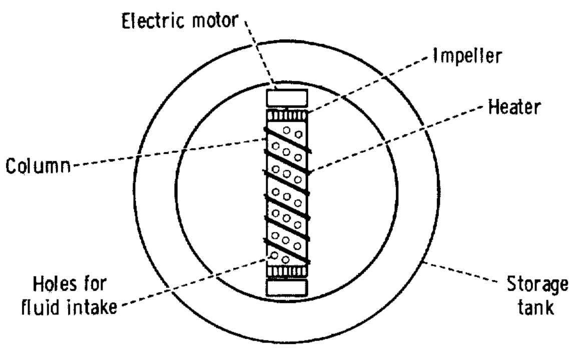


FIGURE 45.—Internal dynamic-heater system.

around and fastened to an internally finned column. The finned column has holes through its lateral surface for fluid intake. Two electric-motor-driven centrifugal impellers, which are located at opposite ends of the heater column, draw fluid through the radial tube holes, force the fluid across the heat-transfer surface, and then expel the fluid radially through the impellers. A dynamic system that involves an external loop does not constrain the location of the heater, motor fan, or motor pump to be within the pressure vessel. The fluid is withdrawn from the pressure vessel by means of a pump, and is routed over an electric heater and is returned to the tank or is expelled to the supply line. An external-loop heater system is shown in figure 46. Improved quantity gaging, homogeneous

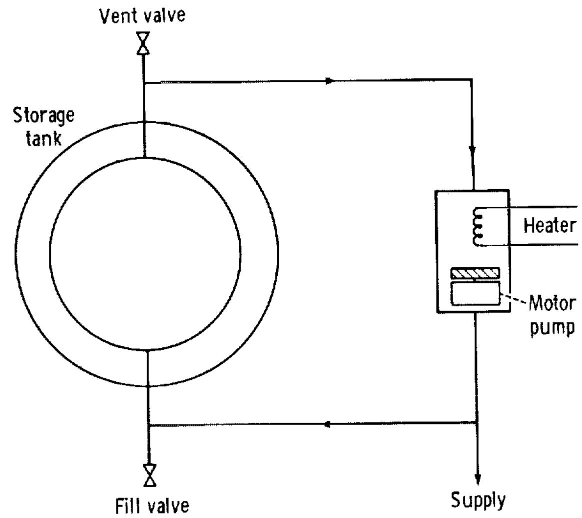


FIGURE 46.—External-loop heater system.

pressurization of the fluid, and operation independent of the gravity environment are advantages of a dynamic system. The reliability of a dynamic system is reduced by the fact that it is dynamic and by the possibility of introducing impurities and excess heat by means of motor operation.

Quantity Measurement

Usually a supercritical storage (single-phase) system involves the use of a coaxial cylindrical capacitor as the sensing element for the measurement of fluid quantity. Operation of this device is based on the Clausius-Mossotti relationship between the density and the dielectric constant of the cryogen. The device senses the dielectric properties of the fluid that is between two capacitor plates. Because (theoretically) the supercritical fluid is homogeneous, the density sample within the capacitor is considered uniform, yielding a corresponding quantity read-out.

Because subcritical storage consists of storing both the liquid and gas phases (therefore, not homogeneous), a simple capacitance probe is not suitable for quantity measurement. A device that has been used for quantity measurement in a subcritical system consists of a cubical matrix arrangement of two electrically isolated lattice-wire grids. The matrix grids are used to measure the fluid capacitance throughout the container volume. When an electric potential is applied to each grid, the two grids act as plates of a capacitor. A matrix can be designed for a desired resolution and tank configuration. Voids can be left in the matrix to fit around baffles or structural members. Because the capacitance of the stored fluid varies as a function of the bulk density of the stored fluid, the capacitance between the grids is measured and translated into an integrated quantity read-out.

A quantity-measuring system that is in the developmental stage, known as the rf gaging system, involves the use of radio-frequency energy. The rf technique involves the use of electromagnetic energy at microwave frequencies to "illuminate" the tank with rf energy, and thus measure the entire quantity of fluid in a tank, regardless of fluid density or its location within the tank. Basically the rf system responds to variations in the average dielectric constant of the stored medium. The dielectric constant is related to fluid density by use of the Clausius-Mossotti relationship. The tanks are operated as rf-resonant cavities; variations in resonant frequencies (caused by dielectric constant changes coupled with different requirements in rf-power absorption by the fluid) in a cavity are used as measurement parameters. The rf system has a sensing antenna that is placed in the tank. The antenna transmits the rf energy throughout the tank and receives the reflected portion of the incident power. Then the reflected signal is converted to a signal that is pulse modulated by measurement-detection circuitry. The resultant signal is integrated to form an analog signal that represents the quantity of fluid that is present in the tank.

Pressure Measurement

Usually, fluid-pressure measurements are obtained by the use of pressure transducers that involve strain gages that are bonded to a diaphragm. The diaphragm deflects in proportion to the applied pressure, and it supplies a millivolt signal to a signal conditioner.

Temperature Measurement

Usually, temperature measurements within cryogenic storage tanks are obtained by means of copper-constantan thermocouples and platinum resistance-sensing elements. The temperature difference between the junction of dissimilar metals produces an electric potential that is a function of the temperature difference. Temperature can be determined by measuring the electromotive force (emf) of the thermocouple, and then converting from emf to difference in temperature between a reference-temperature region and the junction at which the temperature is being measured. The electrical resistivity of conductors and semiconductors is affected directly by temperature. This property is the basis for resistance-sensing elements. Material properties that are desirable include an electrical resistance that varies linearly as a function of temperature, a high sensitivity (great change in resistance as a consequence of small change in temperature), and a time-stable electrical resistance.

Signal Conditioners

Signal conditioners are used in conjunction with the temperature-, pressure-, and quantity-sensing elements. Signal conditioners generate the read-out signal. Usually the signal conditioner amplifies the voltage that is developed from the sensing elements, and supplies a 0- to 5-volt dc output signal that is linearly proportional to the measured parameters. Parameters that are not linear are compensated for in the signal conditioner.

Pressure Switch

Pressure switches are used to energize and deenergize the electrical heater system at predetermined pressures. The pressure switch senses the tank pressure and actuates a toggle mechanism that controls and actuates the heater circuit.

Burst Disk

The burst disk is a thin metallic sheet that is designed to burst when the contained pressure exceeds a preset value. A burst disk is a "one-shot" device that vents the tank pressure until environmental pressure is achieved. Burst disks can be classified as shear bursters or flower bursters. In the shear burster, the disk is flat and fails in shear as it deflects against a sharp-edged holder. Generally this type of device is used for high-pressure applications. For lower pressures, the flower burster is commonly used. The flower-burster disk is contoured and tears from the center to the holder (creating a flower configuration). In the case of vacuum-insulated double-walled cryogenic storage vessels, usually the burst disk is welded into the wall of the outer shell. In the event of a loss of vacuum integrity (for example, leakage from the pressure vessel), the pressure in the annular volume will rise rapidly and eventually will rupture the burst disk. Thus the direction and location of the vented cryogen can be controlled.

Valves

The CGSS includes the following valves for use in the control of fluid flow: fill, vent, pressure relief, supply, and check. The fill valve is used during the filling of the system and is operated either remotely or manually. If it is operated remotely, a solenoid valve is used. Solenoid valves are operated electrically, and usually have a manual override so that the valve can be opened or closed manually. If the valve is operated manually, a quick-disconnect valve equipped with an integral check valve to prevent backflow is used, or a simple hand-operated cryogenic valve is used. After filling, a seal cap is screwed on the fill coupling as a secondary leak seal and cover. The vent valve is used during system filling to facilitate vapor venting, and is operated remotely or manually, similar to the fill valve. The pressure-relief valve prevents the system pressure from rising above a maximum level by allowing the fluid to escape until the pressure is decreased to a safe level, at which time the valve closes. Usually this valve is a mechanical type of valve and the spring tension is adjusted according to the pressure-relief requirements. The supply valve serves as the control for fluid flow during operation of the system. Generally this valve is of the solenoid type. The check valve, which is installed in a fluid line, is used to prevent backflow of the fluid. The valve consists of a spring-loaded poppet that seals against a seat.

Ion Pumps

Ion pumps are used to monitor the vacuum-annulus pressure and are capable of removing small amounts of gases which result from material outgassing. The capacities of the ion pumps that are used vary from 0.2 to 1.0 liter per second. The electrical current that is needed by the ion pump during operation indicates the pressure within the vacuum annulus by correlation with a calibration curve (current plotted as a function of pressure).

Fluid Lines

Fluid-line sizes are dictated by manufacturing considerations and flow requirements. The choice of fluid-line material is governed by cryogen compatibility, joinability, and heat-conduction characteristics.

Filters

One type of filter that is used consists of multiple stainless-steel disks that are stacked to achieve a long flow path. The disk elements are capable of retaining fibers and other contaminants that exceed specific dimensions which are correlated to the disk pore size.

Electrical Connectors

The electrical connectors are sealed hermetically and must be capable of withstanding system pressures and temperatures. The connector pins are sealed in a ceramic material that has the same coefficient of thermal expansion as the connector shell and pin material.

Electrical Leads

The electrical leads that are used for power input and sensor signal output usually consist of conductors surrounded by magnesium oxide insulation and encased in a metallic sheath.

Tank Mounting

After the tank is fabricated and assembled, it must be supported suitably and mounted in the spacecraft. Specific mounting arrangements are dependent on the system design and the spacecraft. Some tanks are supported by their girth ring only, and other tanks may have a mount that encompasses and supports the vessel at various areas or points. Lightweight high-strength metals (such as titanium, aluminum, and magnesium) are favored as a structural-mount material. In the event the system will be exposed to stringent vibration requirements, vibration dampers may be placed between the tank and the spacecraft.

Radiation Shields

A radiation shield is an important component of a CGSS. Two types of radiation shields are discussed in this section.

Discrete shields.—Low-emissivity radiation shields mounted within the dewar vacuum annulus are called discrete shields. Usually, the shields are positioned and mounted by the use of a minimum number of supports. These supports have low thermal conductance. Heat transfer between surfaces in discrete-shielded vessels is composed of solid conduction through the supports and interconnecting lines, conduction by the residual gas, and thermal radiation. Discrete radiation shields have advantages of fabrication, assembly, reproducibility, analytical prediction, vacuum acquisition, and long shelf life. The major disadvantage of discrete radiation shields is that minor surface conditions of the shields have considerable effect on the emissivity values. Because some of the low-emissivity materials will degrade quickly because of oxidation when exposed to air, it is desirable that the tank be assembled in an inert atmosphere or be assembled quickly if exposed to the normal atmosphere.

Vapor-cooled shields. — Discrete shields that have the capability of being cooled by the effluent fluid from the tank are called vapor-cooled shields. Fluid issuing from the pressure vessel, either from venting or usage, is routed through tubing that is attached to the shield. The vapor-cooled shield acts as a heat exchanger between the fluid and the shield, and the fluid picks up heat energy from the shield and carries it out of the system.

Plating Processes

Plating involves the deposition of a film of metal onto the tank pressure vessel, outer shell, and discrete-shield surfaces that are exposed to the vacuum annulus. Plating results in a low-emissivity surface that is used to limit radiative heat transfer; also, plating limits outgassing in a static vacuum environment. Usually, surface plating of cryogenic-tank components is accomplished by means of electroplating and vapor deposition.

Electroplating involves the production and deposition of metallic coatings on the tank parts by means of an electrochemical process. The metal coating is produced by the passage of an electric current through an ionic solution, causing a separation of metal ions at one of the electrodes. Two conducting electrodes are connected to the poles of a suitable source of electromotive force, and the electrodes are placed in an electrolytic solution that is capable of conducting an electric current. When the electromotive force is applied to the electrodes, a current flows through the electrolyte, and because of the voltage difference between the two electrodes, the ions are displaced in the solution. The ions migrate to the electrodes, where they are discharged in the form of a metallic coating. Vapor-phase deposition includes several different processes; however, usually the evaporation and condensation of metals under vacuum conditions is used. A major criterion for the successful use of the vacuum-evaporation process is that the part to be plated must have a smooth surface. A smooth surface can be achieved by the application of a coat of either an epoxy or polyimide varnish that is allowed to level out to a high-gloss surface. Prior to plating, the coatings must be processed to remove any volatile materials. Aluminum, copper, gold, and silver are the customary materials that are plated onto the shield substratum material for low-emissivity shield surfaces.

Vacuum Insulation

A critical requirement of a cryogenic dewar system is the vacuum integrity within the annulus. If vacuum integrity is poor, the dewar will have a high heat leak because of excessive gas conduction of heat that is transferred into the stored cryogen. Then the system will be forced to vent fluid overboard if the demand is not sufficient to use the quantity that results from the increased expulsion rate. All materials contain residual gases which may be released when the material is exposed to a vacuum. The volume of gas released by a material will vary depending upon the material characteristics. To obtain and maintain a high static vacuum, the materials must be processed properly to remove the residual gases. During the evacuation process, usually the dewar is heated, which forces more gases to be emitted and removed from the annulus. To prevent additional outgassing from degrading the vacuum after processing, it is a normal practice to use an absorbent (getter) or vacuum ion pump to remove the gases that are released after the dewar is sealed. Careful selection of materials and vacuum processing can result in a reduction of residual gases that may be outgassed. Complete elimination is impossible. For effective thermal protection, a pressure level of 5×10^{-5} torr or lower is required.

Material Insulation

Of the numerous insulation techniques that are used, the powder and laminar insulations and the discrete shields and vapor-cooled shields have received the most attention. Usually, the insulation material is placed within the dewar vacuum annulus; however, in some cases the laminar insulation is placed externally.

Powder insulations.—This type of insulation consists of finely divided solid materials that have low thermal conductivity. Generally the average density of the powder is low, so that there is a relatively small ratio of solid material to gas-filled spaces between the particles. Heat is transferred within the insulation by means of conduction through the solid particles, conduction and convection through the interstitial gas, and radiative transfer through the partially thermally transparent powder and from particle to particle. Settling or packing of the particles is a problem that may be encountered in the use of powder insulations. Vibrations and movements during usage may break down the powder particles, causing them to come into closer contact with each other and resulting in an increase in solid thermal conductivity. Perlite, silica aerogels, charcoal, diatomaceous earth, and calcium silicate are examples of powder insulations.

Laminar insulation.—This type of insulation consists of alternate layers of radiation-shielding material and low-conductivity spacing material. A common laminar radiation shield is made of aluminum foil, which serves as the reflective shield material. The shields are separated from each other by thin layers of such low-conductivity materials as Fiberglas paper or cloth. Aluminized Mylar, another type of laminar insulation, is made of single sheets that have the reflective material on one side and the low-conductivity material on the opposite side. Thus separate layers of shields and spacers are eliminated. The effectiveness of laminar insulation depends upon the emissivity of the shields and upon thermal isolation between the shields and the number of layers that can be applied in a given space. Usually the layers are spaced loosely or are crinkled purposely to minimize contact between layers. The application of laminar insulation to other than flat or cylindrical surfaces is critical. It is difficult to achieve ideal performance, because the spacing of layers generally is not uniform and allowances must be made for lines, leads, and other objects that interfere with the application. The normal method of applying laminar insulation consists of wrapping, overlapping, applying gore sections, taping, sewing, and heat sealing. Evacuation of the annular space is difficult because of the many surfaces of material and the large quantities of residual gases which must be pumped out of the annulus.

Nonmetallic Materials

Considerations for the selection of nonmetallic materials include such parameters as temperature effectivity, cryogen compatibility, cleanliness, maintainability, and microbial resistivity. Frequently, nonmetallic materials that have low thermal conductivity are used to provide needed support and partial thermal isolation. A vivid illustration of the importance of cryogen compatibility is given in the "Post-Apollo 13 Redesign" section in chapter 2. The incompatibility of Teflon, a nonmetallic material, in the presence of high-pressure oxygen and a potential ignition source contributed to the failure of the Apollo 13 CGSS oxygen tanks.

3 Cryogenic Gas Storage Systems Design and Use on Advanced Manned Missions

Many scientific advances in cryogenics have occurred since Louis Cailletet and Raoul Pictet succeeded in liquefying oxygen and presented their papers on the subject to the French Academy of Sciences. In chapter 1, a history of the early years of cryogenics development was given. This account included an acknowledgment of some early systems efforts and technology advances in the field. These early efforts and advances provided the technology base from which initial spacecraft cryogenic gas storage systems were developed; chapter 2 included a discussion of cryogenic gas storage systems that were designed and built for the Gemini and Apollo spacecraft. However, before these spacecraft cryogenic gas storage systems could be designed, developed, produced, and flown, an orderly sequence of engineering efforts was required (hereafter referred to as the systems-engineering approach and defined in part 1). Therefore, the basic purpose of chapter 3 is to present and discuss the basic principles and implications of the systems-engineering approach, not just as it applies to the Gemini and Apollo cryogenic gas storage systems, but as it applies to any spacecraft cryogenic gas storage system (CGSS). Information is given on some of the possible future space missions and the technology advances that are required. Specific systems-engineering cases will be presented. These cases were chosen because they currently are in the preliminary analysis and design stages. Their use as examples will provide an accounting of current activity and will illustrate the importance and place of these steps in the evolution of a spacecraft CGSS.

Part 2 of this chapter is a somewhat detailed description of the evolution of aerospace CGSS design. The definition of the necessary performance and configuration of a cryogenic storage system arises from a determination of its intended use, through guidelines definition to consideration of the materials requirements and related disciplines. After optimization in these areas, the best that is available or that can be made available (within schedule and cost) may be reduced to detailed design, manufacturing drawings, and so forth. The equivalent labels to the "intended use, guidelines, and scientific disciplines" just mentioned are, respectively, flight objectives and desires, conceptual reference missions or design reference missions, and the thermal and thermodynamic disciplines. Several design reference missions are proposed in part 2 in response to "intended use," which ranges from Earth-orbital and lunar-orbital shuttles to early Mars missions that involve landings. A sample systems-engineering approach is discussed and selected cases are carried through detailed engineering analysis to a determination of storage pressures, cryogen quantities, and flow rates.

The quantity and performance requirements of cryogenic systems in conjunction with far-future missions are given in part 3, which is a less detailed, preliminary systems-engineering approach to the definition of electrical power system and life-support cryogenic-fluid-quantity requirements for manned missions in the somewhat more distant future. A discussion of some of the problems that might be encountered in efforts to store cryogenic fluids on or in the lunar surface is included. The use of these cryogens for power generation and metabolic-gas supply in conjunction with a proposed manned lunar farm and laboratory is discussed also. The far-future-missions part of this chapter is divided into two major sections. As mentioned previously, the first section is a discussion of the use of cryogens in a manned lunar laboratory and farm. A general summary of the problems that are associated with the storage of cryogens on a nonterrestrial surface is included. The second section is a summary of the metabolic use of cryogenic oxygen on advanced lunar and planetary missions. Metabolic-gas system requirements are emphasized more than power-generation requirements, and consideration is given to the effects of propulsion-system selection on cryogenic-gas-storage requirements. Generally the discussion of each planetary mission is divided into two areas of consideration: a discussion of the general mission configuration and goals and a discussion of the cryogenic gas storage systems, manning, and planetary-surface stay times (when applicable).

Since the Apollo 11 mission, the perspective regarding post-Apollo missions has been altered. Missions that had been regarded as distant may now be regarded as imminent. The development of a manned lunar base and a manned space station are significant near-future milestones. It is at this point in time and in technology development that the concepts of near-future missions and far-future missions are separated. Herein, far-future missions are conceived as more advanced lunar missions, such as the moonlab, and rudimentary CGSS requirement profiles for several planetary missions. There has been no attempt to delineate rigorously between near-future and far-future missions.

PART 1—THE SYSTEMS-ENGINEERING APPROACH TO SPACECRAFT CRYOGENIC GAS STORAGE SYSTEM DESIGN

A system may be defined as a collection of components united by definite interactions or interdependencies created so that a definite function can be performed. Systems engineering is the process of consideration of each component and its interaction and interdependency with each of the other components that constitute the system in an effort to optimize the system. The Apollo spacecraft is an example of a very complex and extensive system. The components of this system have been united to perform the definite function of fulfilling the mission objectives of the Apollo Program. These components are referred to as subsystems, and include subsystems for communications, propulsion, navigation, guidance and control, electrical power, structures, and life support. A systems-engineering approach was required in order to optimize these components and the functions that they perform for the overall system. Any one subsystem such as communications, propulsion, or electrical power may or may not be optimum in itself, but may sacrifice its individual best performance for the good of the system (the spacecraft) and its intended function.

The discussion of systems engineering presented in this chapter is intended to serve as an example, and, as such, does not present all of the required methods and areas of consideration. The discussion involves the manner in which it is applied to the design and evolution of a spacecraft CGSS. This same type of approach is used commonly in many other areas of endeavor, and the items to be presented here are representative of those that are used in many of the areas.

An initial mission concept is determined that includes the basic guidelines, ground rules, and constraints necessary for definition of the proposed mission. Basic information such as the primary mission objectives, mission duration, crew size, schedule, the number and type of spacecraft involved, and spacecraft size is part of this initial concept. This information is preliminary in nature, and, therefore, is not detailed or exact. The purpose of the subsequent steps of systems engineering is to define, verify, modify, or add to the original mission concept.

After an initial mission concept has been determined, it is possible to propose a conceptual reference mission (CRM). The term conceptual reference mission has been coined to prevent confusion with the design reference mission (DRM). A CRM profile is comprised of only such factors as the proposed mission goals, mission duration, and crew size. A CRM may be contrived in general terms and without the extensive manpower and funding that are necessary for DRM definition. Generally, a CRM would be the precursor of a DRM; however, not every CRM becomes a DRM. Basically the two concepts differ in the level of detail that has been established for a particular mission. For example, a lunar mission is a good example of a DRM because operational details such as crew size, cabin volume, gas-leakage rates, mission duration, and velocity requirements can be made definite based on current information and the state of the art. A mission to Mercury is a good example of a CRM because the critical operational details cannot be defined accurately based on the current state of the cryogenics art.

During the CRM phase, a preliminary assessment is made of such items as potential cryogenic-fluid user systems (life support, propulsion, and power), available technology base, required technical advances, cost, and schedule. Once this assesment has been made, it is possible to determine the basic feasibility of the CRM. Then several conceptual reference missions may be compared, and an optimization trade-off between them may be conducted. A determination may be made as to which conceptual reference missions should be dropped from further consideration and which ones should become design reference missions.

Those missions that are chosen to become design reference missions are passed to the next tier of definition and analysis. Information transferral is set up between subsystems that affect each other directly. Then an iterative process of analysis, assessment, and definition is conducted. A constraint placed on one subsystem (or component) has a subsequent effect on other subsystems. For cryogenic systems, these iterations include determination of the cryogen quantities that are required, volume limitations, weight, thermal performance, operational modes, thermodynamic effects, interface requirements, and cost. The interactions of these items as a subsystem and the effect of these items on other subsystems must be evaluated. Finally, trade-off optimizations are conducted to assess items such as system weight as a function of thermal performance, supercritical storage compared with subcritical storage, and multiple-tank configurations compared with single-tank configurations. As an example, a dewar supported by a single-point contact, refrigerated, and externally insulated would be ideal from the standpoint of thermal performance. However, the single-point support could not withstand the launch environment; a refrigeration unit would impose additional power, volume, and weight requirements; and the external insulation would impose additional volume and weight requirements. The trade-offs are made to determine which items contribute the most to overall subsystem efficiency in the performance of the intended function. Once this determination has been made, the net result may be that no given item is optimum in itself. However, the total subsystem performance is better than it would have been if each component had been optimized individually without relation to the others. Because the subsystem subsequently functions as a unit, the systems-engineering objective has been accomplished. Detailed examples of some of the areas of consideration mentioned in this discussion are presented in parts 2 and 3 of this chapter.

Development, production, and subsequent use on a mission are the logical concluding steps in this sequence of events. The repeated successful performance of the many subsystems that have been used in space flights is adequate testimony to the value of systems engineering.

PART 2—ORBITAL SHUTTLES AND EARLY NONTERRESTRIAL LANDERS

Subsequent to the Apollo Program, the Skylab Program and the continued exploration of space assumed definitive proportions in early 1970. Part 2 of chapter 3 is a discussion of the evolution of cryogenic gas storage systems for use on the next generation of lunar vehicles and on the early planetary orbital and landing vehicles. These proposed missions are described only to the extent that is necessary for the definition of the ultimate size and required performance of the cryogenic systems. After such definition, preliminary quantification of the system requirements is possible and is given.

Design Reference Mission

In an attempt to ascertain subsystem requirements for a particular mission, it is helpful to describe a DRM in detail. The use of the DRM results in the establishment of base-line requirements and is helpful for detailed study definition. Design reference missions were used in various forms in the planning of most flight-systems development, and the concept is basic to the systems-engineering approach. Generally the DRM requirements are severe in relation to the planned or probable missions. This results in an indirect redundancy because the study results in the definition of more stringent designs than are necessary in the actual situation. Then the finished product is applicable to both the DRM and a variety of similar mission objectives.

This consideration will involve design reference missions that have been approved in scope by NASA, but which have not yet been approved in great detail. Some of the required detail is supplied, and the gross requirements and weights for several applicable cryogenic gas storage systems are presented parametrically. The feedthrough from philosophy and guidelines to design-requirements definition will be shown.

At the present time, three general categories of advanced missions may be defined, as shown in the following list.

(1) Earth-orbital missions that require a surface-to-orbit shuttle

(2) Lunar-orbital missions that require an Earth-to-lunar-orbit shuttle (may require a lunar lander)

(3) Mars-orbital missions (may require a Mars lander)

Systems-Engineering Philosophy

From a programmatic and systems-engineering viewpoint, a complete DRM involves the entire spectrum of mission criteria, from navigational parameters, thrust levels, and structural design to consumables requirements and waste disposal. However, a DRM usually is segmented by discipline for ease of assessment, and, in this section, design reference missions will be analyzed only in terms of the cryogenic-fluid storage and quantity requirements. These requirements are determined from complex mission planning, and, as such, they are dependent upon a variety of operational considerations. For example, a requirement to orient solar panels for full sunlight may result in the need for less fuel-cell reactant because of decreased power requirements, and may result in the need for more reaction-control propellant because of a requirement for increased maneuvering. Based upon statements of intent regarding the mission (such as the solar-cell orientation just mentioned), an individual case may be qualified.

Case by case, the detailed fluid-quantity or reactant-quantity requirements and the ultimate system design may be evolved. The following statements of intent for the selected design reference missions are applicable for the discussion of near-future missions.

(1) Space base or station

(a) Provide for 10 to 12 crewmen on 6-month tours of duty

(b) Design for a 10-year life including maintenance and parts replacement

(c) Provide the capability to translate the space base to lunar or Mars orbit from Earth orbit

(2) Shuttle

(a) Provide a totally reusable vehicle that can ferry 10 to 12 passengers and that can resupply the Earth-orbital space base

(b) Provide for extension of shuttle capability to the vicinity of the Moon

(3) Lunar lander

(a) Provide a reusable vehicle for round-trip lunar-surface excursions from a space base in lunar orbit

(b) Provide for a 1-week stay time for six men

(4) Mars lander

(a) Provide a reusable vehicle for round-trip Mars-surface excursions from a space base in Mars orbit

(b) Provide for a 6-week stay time for six men

The specific design reference missions that are to be considered will be labeled and referred to as shown in the following chart.

DRM	Description
IA	Earth-orbital space base/station
IB	Shuttle to Earth-orbital space base/station
IIA	Lunar-orbital space base
IIB	Shuttle from Earth-orbital to lunar-orbital space base
IIC	Lunar lander from lunar-orbital space base
IIIA	Mars-orbital space base
IIIB	Mars lander from Mars-orbital space base

System Guidelines

After the mission specifics such as duration and crew size have been defined, it is necessary to define the next level of requirements. This next level begins with the determination of cryogenic-fluid requirements. The life-support requirement consists of leakage and oxygen-consumption considerations. Oxygen and nitrogen leakages are plotted as a function of cabin volume in figures 47 and 48. The Apollo cabin leak rate is very low, but the leak rate per unit of cabin volume must be reduced on future vehicles so that the waste of large quantities of gas into space can be prevented. The leak

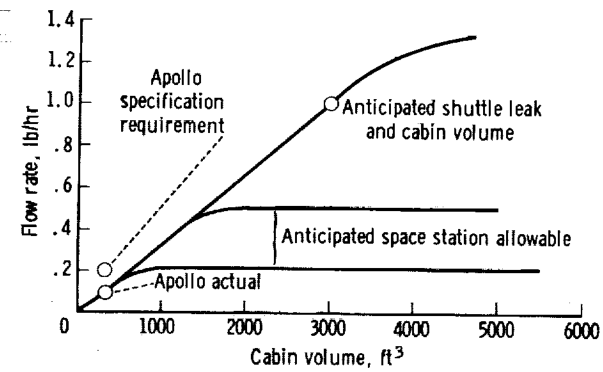


FIGURE 47.—Oxygen leak rates plotted as a function of cabin volume.

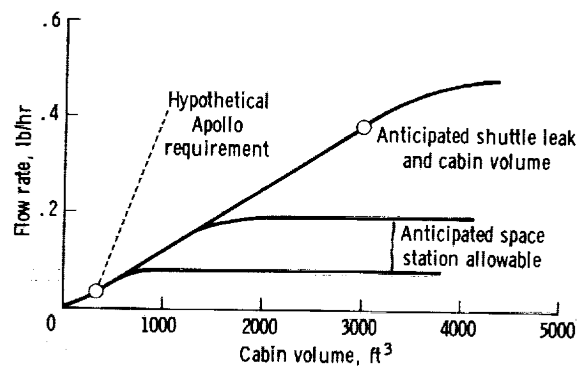


FIGURE 48.—Nitrogen leak rates plotted as a function of cabin volume.

rates from figures 47 and 48 are included in the calculations that determine table 25, which contains the details that are required for continuation of the system description. Two different mission durations are shown for DRM IB (shuttle to Earth-orbital space base), IIA (lunar-orbital space base), and IIIB (Mars lander from Mars-orbital space base). One of these mission times will be dropped as the parameters under consideration proliferate; initially, both times are included for information purposes if analyses separate from this report are to be performed. These leakage rates are realistic relative to experience, but they lend a conservative bias to the quantity estimates because future missions require lower leak rates. This is evident when the inadvisability of transporting fluids into deep space and subsequently wasting them is examined.

TABLE 25.—*Metabolic-Gas Quantity Summary*

Parameter	Design reference mission						
	IA	IB	IIA	IIB	IIC	IIIA	IIIB
Total number of crewmen and passengers	12	12	12	12	6	12	6
Resupply interval, maximum, hr	4320	(a)	(a)	(a)	168	17 280	1008
Low-range interval,[b] hr	(a)	168	4320	168	(a)	(a)	(a)
High-range interval,[b] hr	(a)	720	8640	720	(a)	(a)	(a)
Metabolic-gas use rate, lb/man-hr	0.083	0.083	0.083	0.083	0.083	0.083	0.083
Cabin leak rate							
O_2, lb/hr	0.5	1.0	0.2	1.0	0.2	<0.2	<0.2
N_2, lb/hr	0.19	0.37	0.07	0.37	0.07	<0.07	<0.07
Total metabolic-gas quantity, lb	4302.7	(a)	(a)	(a)	83.66	17 210.8	501.98
Low-range interval[b]	(a)	167.3	4320.7	167.3	(a)	(a)	(a)
High-range interval[b]	(a)	717.1	8605.4	717.1	(a)	(a)	(a)
Total gas leakage							
O_2, lb	2160	(a)	(a)	(a)	33.6	<3456	201.6
Low-range interval[b]	(a)	168	864	168	(a)	(a)	(a)
High-range interval[b]	(a)	720	1728	720	(a)	(a)	(a)
N_2, lb	820.8	(a)	(a)	(a)	11.76	<1209.6	70.6
Low-range interval[b]	(a)	62.2	302.4	62.2	(a)	(a)	(a)
High-range interval[b]	(a)	266.4	604.8	266.4	(a)	(a)	(a)
Metabolic-gas subtotal[c]							
O_2, lb	6462.7	(a)	(a)	(a)	117.26	<20 666.8	703.6
Low-range interval[b]	(a)	335.3	5166.7	335.3	(a)	(a)	(a)
High-range interval[b]	(a)	1437.1	10 333.4	1437.1	(a)	(a)	(a)
N_2, lb	820.8	(a)	(a)	(a)	11.76	<1209.6	70.6
Low-range interval[b]	(a)	62.2	302.4	62.2	(a)	(a)	(a)
High-range interval[b]	(a)	266.4	604.8	266.4	(a)	(a)	(a)
Metabolic-gas quantity needed for complete repressurization							
O_2, lb	1610	178.6	1610	178.6	(d)	1610	(d)
N_2, lb	573.7	63.6	573.7	63.6	(e)	573.7	(e)
Quantity of O_2 needed for airlock,[f] lb	4.3×12	4.3×2	4.3×12	4.3×2	4.3×4	4.3×48	4.3×8
Repressurization-gas subtotal[g]							
O_2, lb	1661.6	187.2	1661.6	187.2	17.2	1816.4	34.4
N_2, lb	573.7	63.6	573.7	63.6	(h)	573.7	(h)
Metabolic-gas total[i]							
O_2, lb	8124.3	(a)	(a)	(a)	134.4	<22 483.2	738
Low-range interval[b]	(a)	522.5	6828.3	522.5	(a)	(a)	(a)
High-range interval[b]	(a)	1624.3	11 995	1624.3	(a)	(a)	(a)
N_2, lb	1394.5	(a)	(a)	(a)	11.8	<1783.2	70.6
Low-range interval[b]	(a)	125.7	876.1	125.7	(a)	(a)	(a)
High-range interval[b]	(a)	330	1178.3	330	(a)	(a)	(a)

[a] Not applicable.
[b] For DRM IB, IIA, and IIB, a low-range resupply interval and a high-range resupply interval were selected; selection of values was based on an inspection of various parameters (principally stay time) of a particular DRM.
[c] Subtotal=total metabolic-gas quantity+total gas leakage.

However, optimization of the cost trade-offs that are involved in thermal-protection improvements compared with fluid venting is beyond accurate analysis at this time. Therefore, it is assumed in this analysis that some leakage occurs.

From the mission specifics and the subsequently definable requirements, it is possible to determine the total cryogen quantities and performance characteristics of the applicable cryogenic gas storage systems. Unlike the Gemini and Apollo spacecraft, cryogenic fluids will be used as propellants in most cases on the next generation of spacecraft. This use of cryogens will be in addition to the traditional roles as fuel-cell reactants and cabin-atmosphere sources. Accordingly one more design iteration should be performed (in the absence of other operational constraints) to determine how best to configure the tankage and dispense these cryogenic fluids. One extreme method of tankage configuration would be to combine all cryogenic fluids into one vessel that has bulkhead compartments which separate the individual fluids.

Another extreme configuration has been referred to as a tank farm, wherein each function and fluid is isolated in separate tanks. However, differences in storage pressure, delivery techniques, and storage duration that are associated with each of the fluid functions will simplify such analyses by the elimination of some of the possible variables. This occurs because of such considerations as differences in inlet purity and pressure to the user systems. Because of the complexity of this consideration and the lack of propellant-requirements definition, no parametric analysis of cryogenic-propellant storage will be attempted. It will be noted in the subsequent discussion that the metabolic requirements alone represent a great need for improvement in storage and delivery techniques. In any event, the analytical techniques described herein may be applied to many CGSS design problems.

System Design

After the philosophy and guidelines that are basic to the system have been developed, the design requirements may be determined. From these requirements, it will be possible to compare the new requirements to the state of the art in the areas of size and thermal protection. This will accomplish a first-look analysis of the systems under discussion and will complete this example of cryogenic-systems engineering.

Power-System Considerations

The power system for the Earth-orbit shuttle will consist of fuel cells and batteries as was the case on the Apollo spacecraft. As shown in figure 49, the anticipated quantity requirements are similar to those of Apollo, and the Apollo fuel-cell technology is available as a development base. Technical advances will be required because the Apollo fuel cells were designed to furnish power for the nominal lunar-landing mission, and the fuel cells were designed for only one mission, whereas the planned extensive reuse of the shuttle necessitates a simple, efficient, and sturdy fuel-cell system that is capable of a long life with little or no maintenance.

Now that the life-support cryogenic-fluid requirements for the subject design reference missions have been discussed, the possibility of cryogenic-fluid usage as a fuel-cell reactant will be considered. In figure 49, fuel-cell reactant requirements are plotted as a function of generated power. This figure is indicative of the very large reactant quanti-

[d] The lander will be supplied by the mother ship.
[e] Emergency repressurization does not involve N_2.
[f] The second number of each entry represents the number of repressurizations that are considered in the calculation.
[g] Subtotal = repressurization-gas quantity + airlock O_2 quantity.
[h] The airlock does not involve the use of N_2 or other diluent gas.
[i] Total = metabolic-gas subtotal + repressurization-gas subtotal.

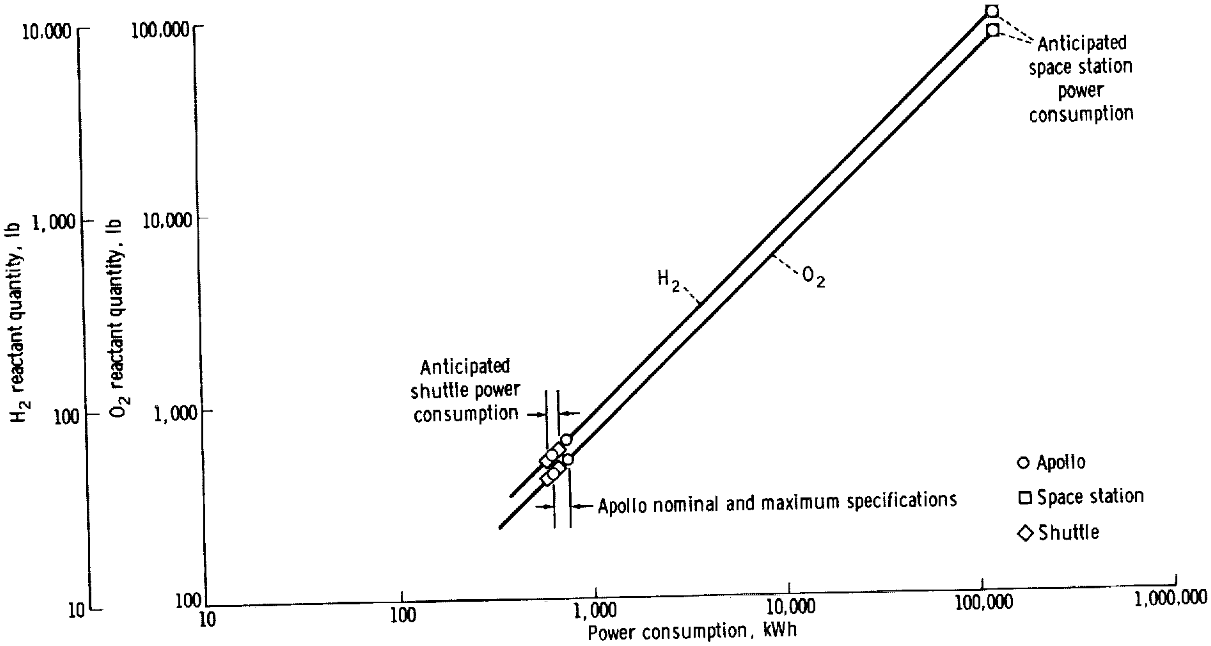


FIGURE 49.—Reactant requirements for a fuel-cell power-supply system.

ties that are required for a power-generation system that operates in ranges applicable to this discussion and that consists of fuel cells only. Accordingly, for long-duration missions that involve large power levels, a fuel-cell power-generation system does not compare favorably on a power-to-weight basis or on a maintenance-interval basis either with a nuclear system or with a solar array/battery system.

A comparison of the fuel-cell cryogenic-reactant weights for an array of possible power systems wherein fuel cells are used in conjunction with other power sources is shown in table 26. A range of cases in which fuel cells would supply 25 to 100 percent of the required power is shown. Again, the longer missions result in excessive reactant weight. Also, the lunar-landing and Mars-landing missions have been sized grossly with respect to power requirements, and the data are given. Obviously the space station or space base cannot provide direct power support for the lander after separation from the orbiting base. However, the landers may contain rechargeable batteries; therefore, no cryogenic fluids are postulated here for power generation on board a lander. If, as seems likely, the landers require cryogenic-fluid storage for metabolic provisioning in the final design, the quantities that are shown in table 26 are valid.

Cryogenic Gas Storage Systems for Near-Future Missions

To size a given vehicle and its subsystems, usually several iterations of comparative analysis are required. The available funding, operational objectives, and available technology are the broad labels for the parameters that are to be balanced. To assist in the preliminary determination of the volumes required for the storage of various quantities of cryogenic fluids, a straight-line plot is shown in figure 50. For oxygen, hydrogen, and nitrogen, the liquid density at 1 atmosphere that is associated with the subject cryogens is multiplied by the appropriate volume extremes. Because the liquids of concern here may be regarded as incompressible, no complex procedures are considered to load the storage vessel with anything other than

TABLE 26.—*Quantity Requirements for Fuel-Cell Reactants*

Application	Extent of fuel-cell participation in power-supply requirements											
	100 percent			75 percent			50 percent			25 percent		
	kWh	O_2, lb	H_2, lb	kWh	O_2, lb	H_2, lb	kWh	O_2, lb	H_2, lb	kWh	O_2, lb	H_2, lb
Shuttle to lunar orbit (Apollo)												
Maximum quantity, lb	739.2	525.2	66.2	554	393.9	49.6	369.6	262.6	33.1	184.8	131.3	16.5
Minimum quantity, lb	621.6	441.6	55.6	466	331.2	41.7	310.8	220.8	27.8	155.4	110.4	13.9
Shuttle to Earth orbit												
Maximum quantity, lb	672	447.4	60.13	504	358.1	45.1	336	238.7	32.7	168.0	119.3	15.03
Minimum quantity, lb	588	417.8	52.62	441	313.3	39.46	294	208.9	26.3	147.0	104.4	13.15
Space station quantity requirement, lb	108 000	73 736	9 664	81 000	57 552	7 248	54 000	38 368	4 832	27 000	19 184	2 416
Space base quantity requirement, lb	432 000	306 942	38 658	324 000	230 267	28 993	216 000	153 471	19 329	108 000	76 736	9 664
Lunar lander quantity requirement,[a] lb	96	68.2	8.59	72	51.1	6.4	48	34.1	4.3	24	17.0	2.1
Mars lander quantity requirement,[a] lb	1 814.4	1 289.1	162.3	1 360.8	966.8	121.8	907.2	644.6	81.2	453.6	322.3	40.6

[a] These quantities are similar to the contingency power requirements, because the space base will not support a lander crew directly, but will carry cryogenic fluids for use on the lander.

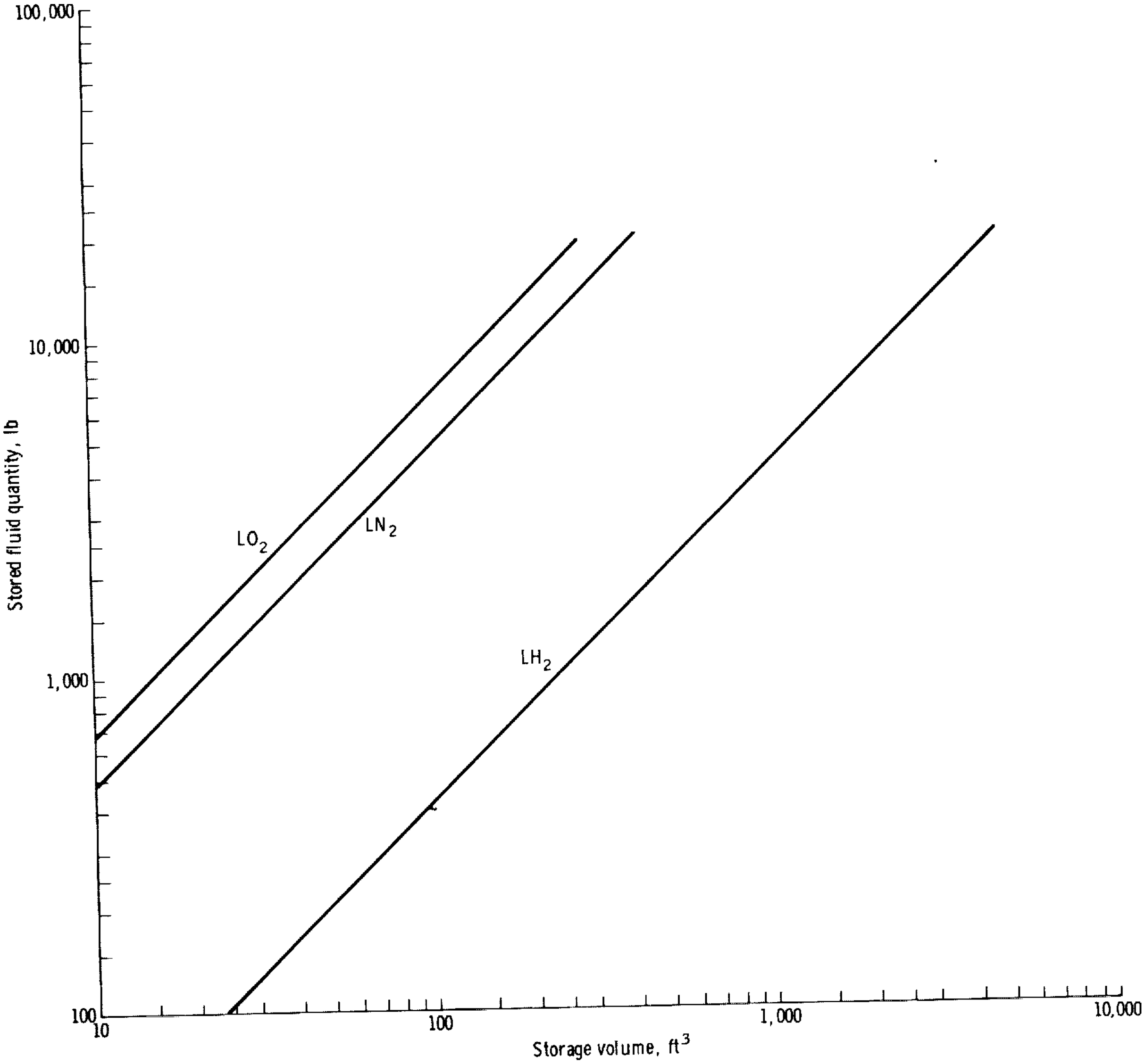


FIGURE 50.—Stored fluid quantity plotted as a function of storage volume for 95-percent fill.

liquid at 1 atmosphere. A 5-percent ullage volume (95 percent by volume fill) is included in figure 49 so that a realistic volume requirement for the bulk fluid will result. In the design of the cryogenic gas storage systems mentioned in preceding chapters, the optimization of vented fluid compared with cost is always a primary consideration. In the case of spacecraft design, the cost parameter is interpreted as dollars and launch weight. To date, it has not been possible to design and build cryogenic gas storage systems that do not vent under normal conditions. On extremely lengthy missions, cryogenic-fluid venting may become an operational necessity. However, the cost in weight and money as a function of payload usually is too great to permit wasteful venting in space. Accordingly, venting is ruled out generally, and the weight optimization must proceed so that the opposing performance and weight requirements can be balanced.

Before proceeding to a more detailed discussion of weight optimization for several cryogenic gas storage systems, it is necessary to detail more exactly the DRM dimensions of duration (or time between resupply) and quantities. As was mentioned previously, electrical power generation by fuel cells only is very unlikely. Conversely, no participation by fuel cells eliminates the cryogenic reactant requirement. In the latter case, metabolic oxygen and nitrogen constitute the only cryogenic-fluid requirement. Accordingly, 0- and 100-percent fuel-cell-participation data are deleted from table 27, whereas 25- and 75-percent fuel-cell-participation data are included. This table is a summary of the quantities that will be needed for the selected design reference missions.

It is now appropriate to discuss some important design details of the CGSS. These design details will include the weight-optimization criteria of storage pressure and thermal protection as determined by equilibrium fluid-expulsion rates.

Storage pressure.—One of the design criteria for storage-pressure selection is minimum flow rate. From fluid-use profiles (which are based upon vehicle operations), it is possible to isolate the lowest flow rate that can be expected during a mission. Once the lowest flow rate that will occur has been anticipated, the storage pressure for the no-venting case can be selected. The first step in this selection process involves a consideration of the variable quantities of heat energy that are required in order to expel a unit mass of fluid.

The heat energy needed to expel a unit mass of fluid from a supercritical constant-pressure cryogenic gas storage system is a minimum during the middle one-third of the mission; the heat-energy requirement rises to a maximum value near the time of depletion. The minimum dQ/dM point determines the highest natural flow rate that will occur provided the heat input (heat leak) remains relatively constant. Flow rate $=$ heat leak$/dQ/dM$, and Q is heat energy and M is fluid mass. The appendix contains useful information in this regard. Unfortunately for the pressure-vessel weight budget, the higher storage pressures and pressure-vessel weights associated with the supercritical mode of storage also result in the lowest natural flow rates. It should be noted that lower natural flow rates reduce the possibility of venting or relieve the thermal-protection requirements. This situation provides the first clear opportunity to perform optimization analyses. The maximum natural flow rates for oxygen, nitrogen, and hydrogen are plotted as a function of heat leak for various pressures in figures 51 to 53. It can be seen from these figures that fluid-orientation devices coupled with subcritical (low) storage pressures could accomplish the same low-flow performance as could supercritical storage. However, highly reliable fluid-orientation devices currently are not flight-qualified, although ground-based tests on some concepts have been favorable.

The flow integral.—When the storage pressures have been selected tentatively from the plots of maximum flow rate as a function of heat leak, it is necessary to examine the possibilities for venting and to initiate the weight trade-offs. Conservatively, the lowest expected flow rates are the metabolic requirements alone. If this metabolic requirement is superimposed upon an array of flow profiles for a matrix of pressures, the potential for vented fluid may be evaluated graphically. As an example, the area under the flow-profile curve and above a horizontal line placed at some specific flow rate should be noted. This area may be integrated to determine the unused (potentially vented) fluid. The natural-flow profiles for several pressures and heat leaks in the temperature regimes that are of interest are shown in figures 54 to 57. Although hydrogen is not needed for metabolic systems, figures 58 and 59 are included for reference in support of any separate analysis that involves hydrogen minimum flow rates for propulsion or for power generation. The use of the hydrogen figures is dependent upon the assumed percentage of fuel-cell participation and the overall power profile. More

TABLE 27.—*Summary of Gas Quantity Requirements*

Design reference mission	Resupply interval, hr	Characteristic											
		Metabolic gas, lb		Fuel-cell-system participation in power requirements				Total requirement					
				For 75-percent power		For 25-percent power		For 75-percent power			For 25-percent power		
		O_2	N_2	O_2	H_2	O_2	H_2	O_2	N_2	H_2	O_2	N_2	H_2
IA	4 320	8 124.3	1 394.5	57 551.6	7 248.6	19 183.9	2 416.1	65 675.9	1 394.5	7 248.3	27 308.2	1 394.5	2 416.1
IB	168	522.5	125.7	358.1	45.1	119.3	15.0	880.6	125.7	45.1	641.8	125.7	15.0
IIA	8 640	1 995.0	1 178.5	115 103.2	14 496.6	38 367.8	4 832.2	127 098.2	1 178.5	14 496.6	50 362.8	1 178.5	4 832.2
IIB	720	1 624.3	330.0	844.1	106.3	281.4	35.4	2 468.4	330.0	106.3	1 905.7	330.0	35.4
IIC	168	134.4	11.8	89.5	11.27	29.8	3.75	223.9	11.76	11.27	164.2	11.8	3.8
IIIA	17 280	<22 483.2	<1 783.3	230 206.4	28 993.2	76 735.6	9 664.4	252 689.6	<1 783.3	28 993.2	99 218.8	<1 783.3	9 664.4
IIIB	1 008	738	70.6	966.8	121.8	322.3	40.6	1 704.7	70.6	121.8	1 060.3	70.6	40.6

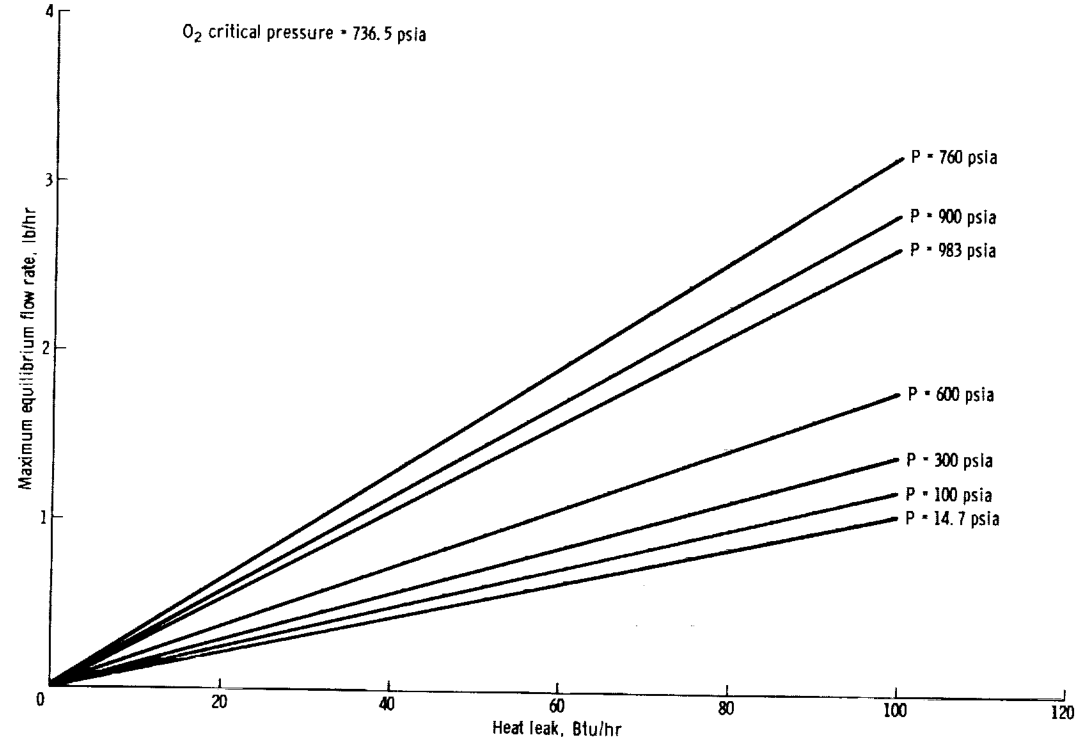


FIGURE 51.—Oxygen maximum equilibrium flow rates at various storage pressures and minimum values of fluid-expulsion heat requirements.

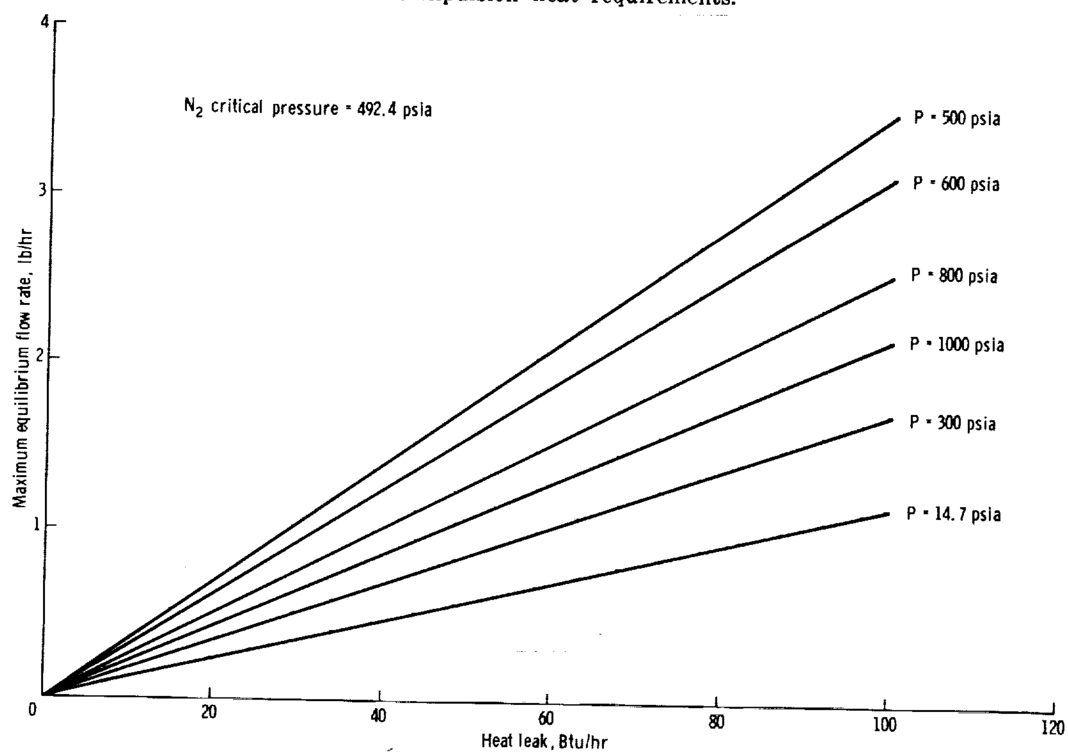


FIGURE 52.—Nitrogen maximum equilibrium rates at various storage pressures and minimum values of fluid-expulsion heat requirements.

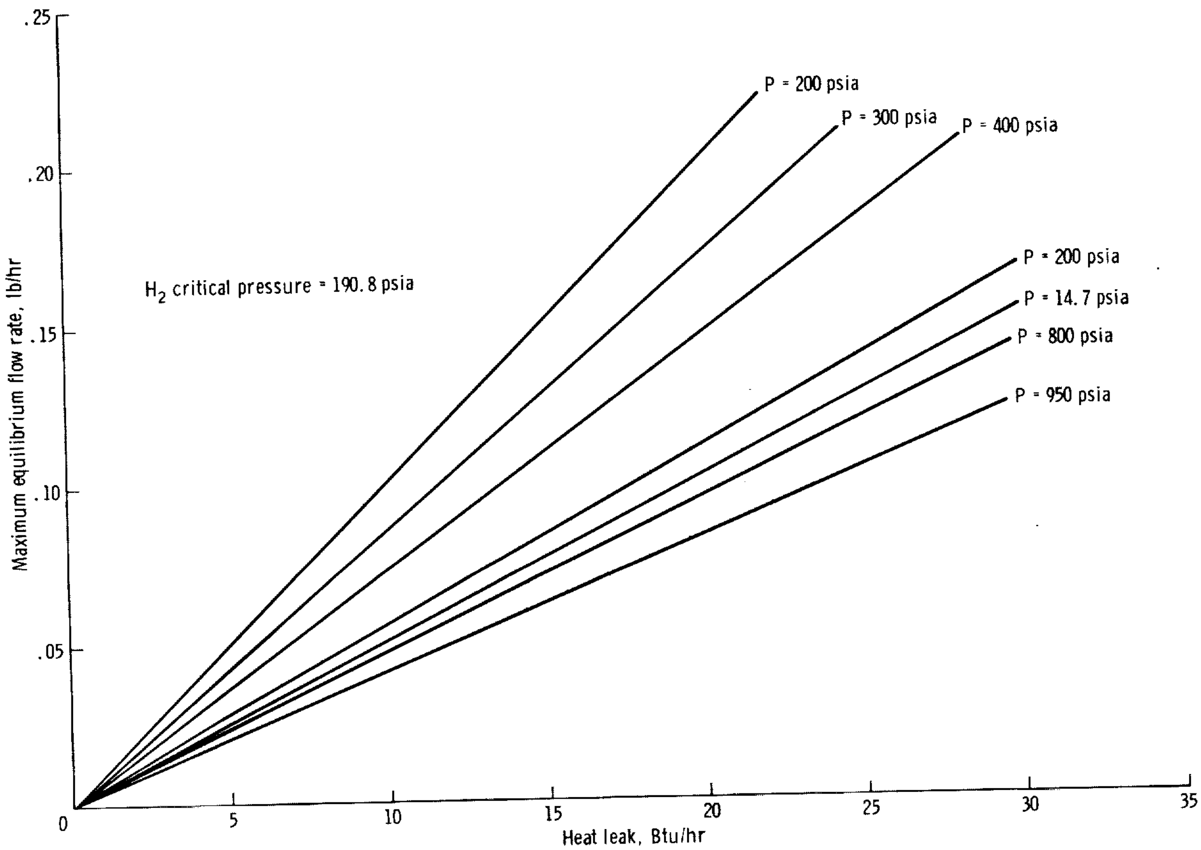


FIGURE 53.—Hydrogen maximum equilibrium flow rates at various storage pressures and minimum values of fluid-expulsion heat requirements.

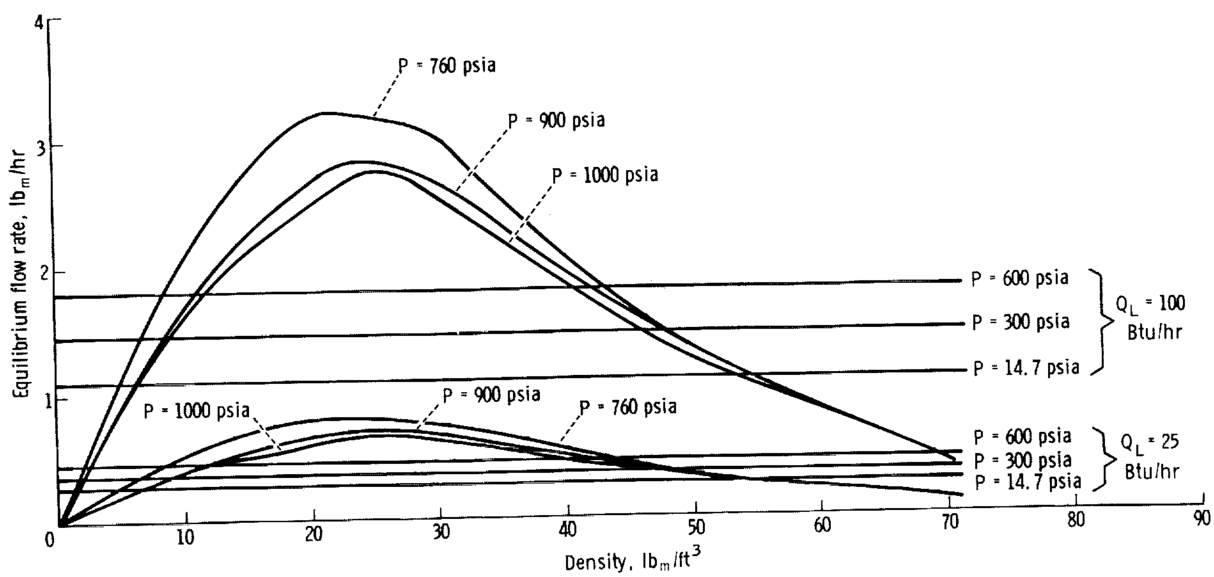


FIGURE 54.—Oxygen flow characteristics for heat leaks of 25 and 100 Btu/hr.

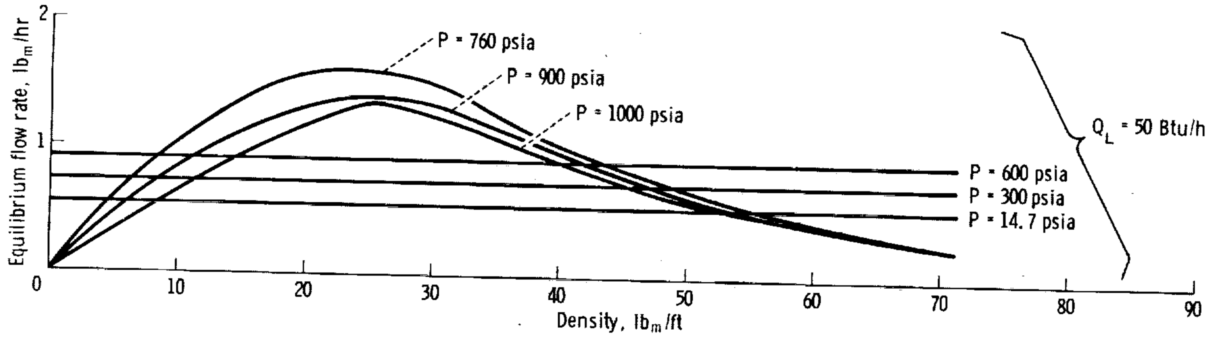


FIGURE 55.—Oxygen flow characteristics for heat leak of 50 Btu/hr.

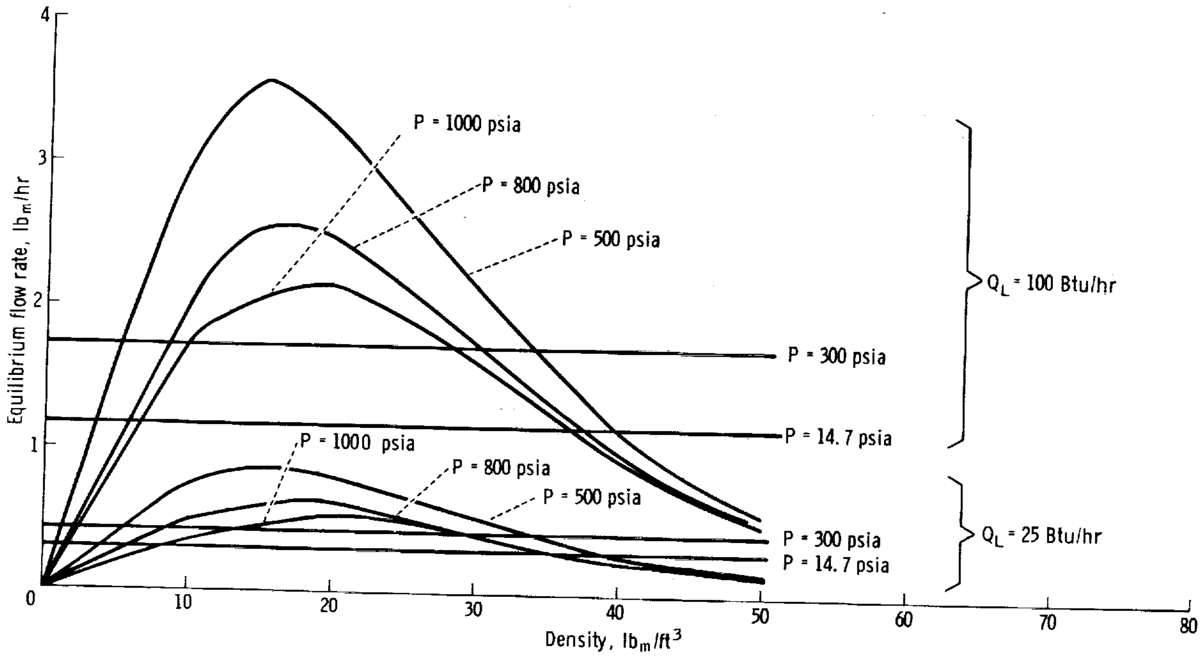


FIGURE 56.—Nitrogen flow characteristics for heat leaks of 25 and 100 Btu/hr.

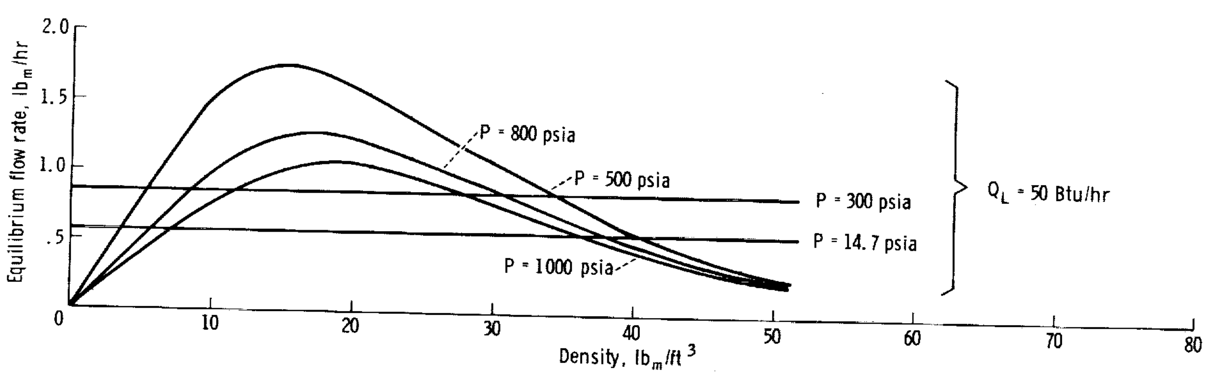


FIGURE 57.—Nitrogen flow characteristics for heat leak of 50 Btu/hr.

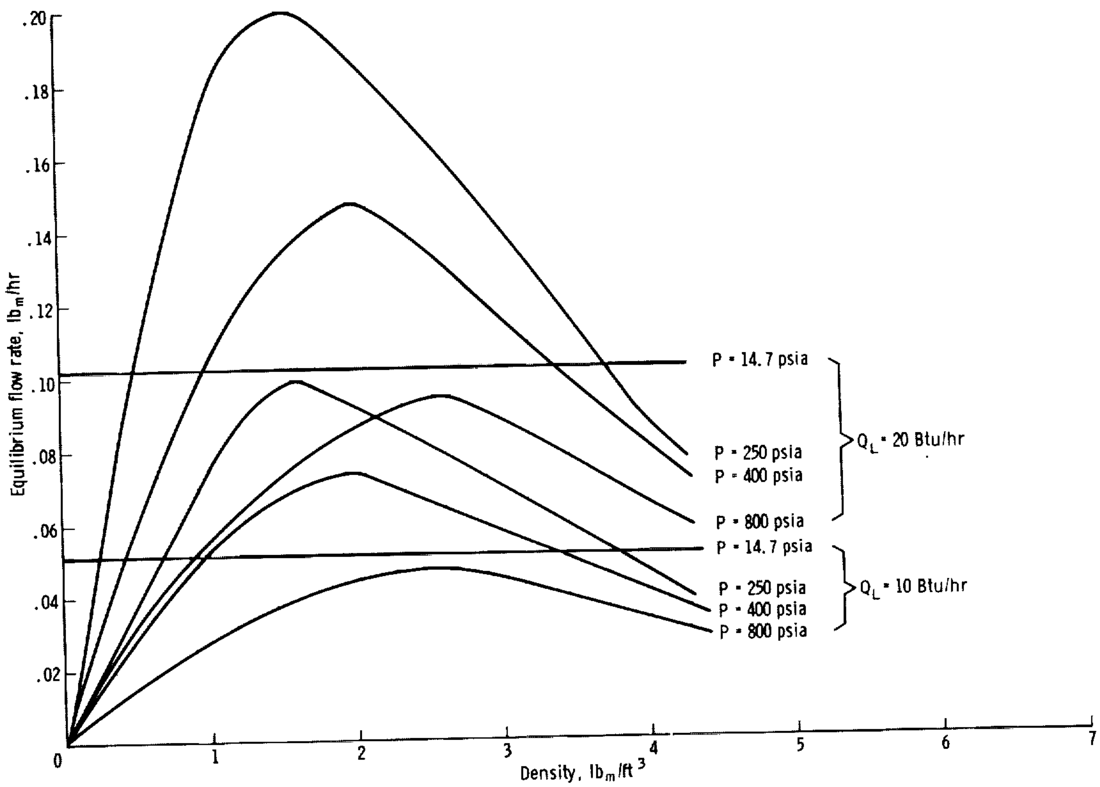


FIGURE 58.—Hydrogen flow characteristics for heat leaks of 10 and 20 Btu/hr.

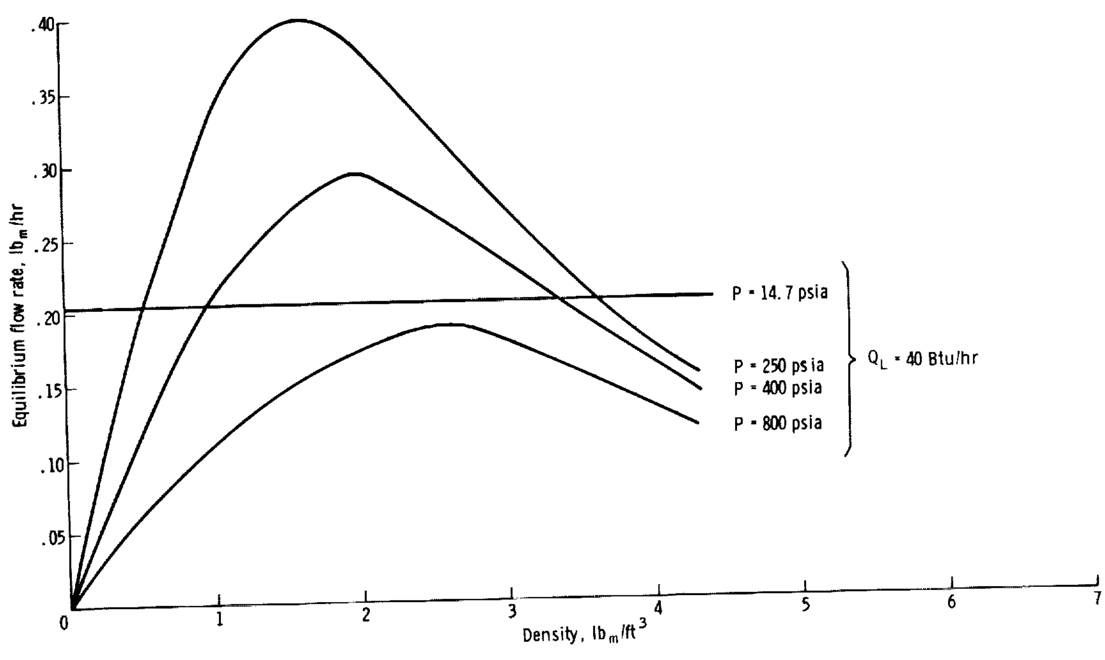


FIGURE 59.—Hydrogen flow characteristics for a heat leak of 40 Btu/hr.

detailed knowledge of the currently inexact operational details eventually will replace these assumptions.

The final major consideration regarding vented-fluid optimization is the comparison of the thermal-protection-scheme weight with the proposed vented-fluid weight. Because the fluid vent rate will decrease as the heat leak decreases, the tendency is to impose constraints that are excessively severe on the thermal design and on the insulation material. This solution to the multiparameter optimization problem should be avoided. Major advances in the state of the art relative to thermal design emerge slowly, and they should be programed only as a last resort. Parametric fluid-vent-rate data are given in figures 60 to 70 for the design reference missions considered here. In each case, the quantity of fluid that is required as a minimum for each DRM may be expressed as an average flow rate, and this flow rate has been compared with the performance of a cryogenic storage vessel that is depleted at equilibrium flow and subjected to a variety of heat leaks and storage pressures. This study was done only for nitrogen in conjunction with oxygen, because the leak rates and mass fraction of nitrogen generally are representative of all of the metabolic diluent gases. In themselves, the metabolic requirements represent a minimum as was stated previously.

By the use of the vent-rate figures, the optimum (although broad) solutions become apparent. The figures and tables presented in the near-future-missions portion of this chapter were intended to be general enough for use with other design reference missions. However, for the subject design reference missions, it may be helpful to use figure 71 to determine the CGSS surface areas that

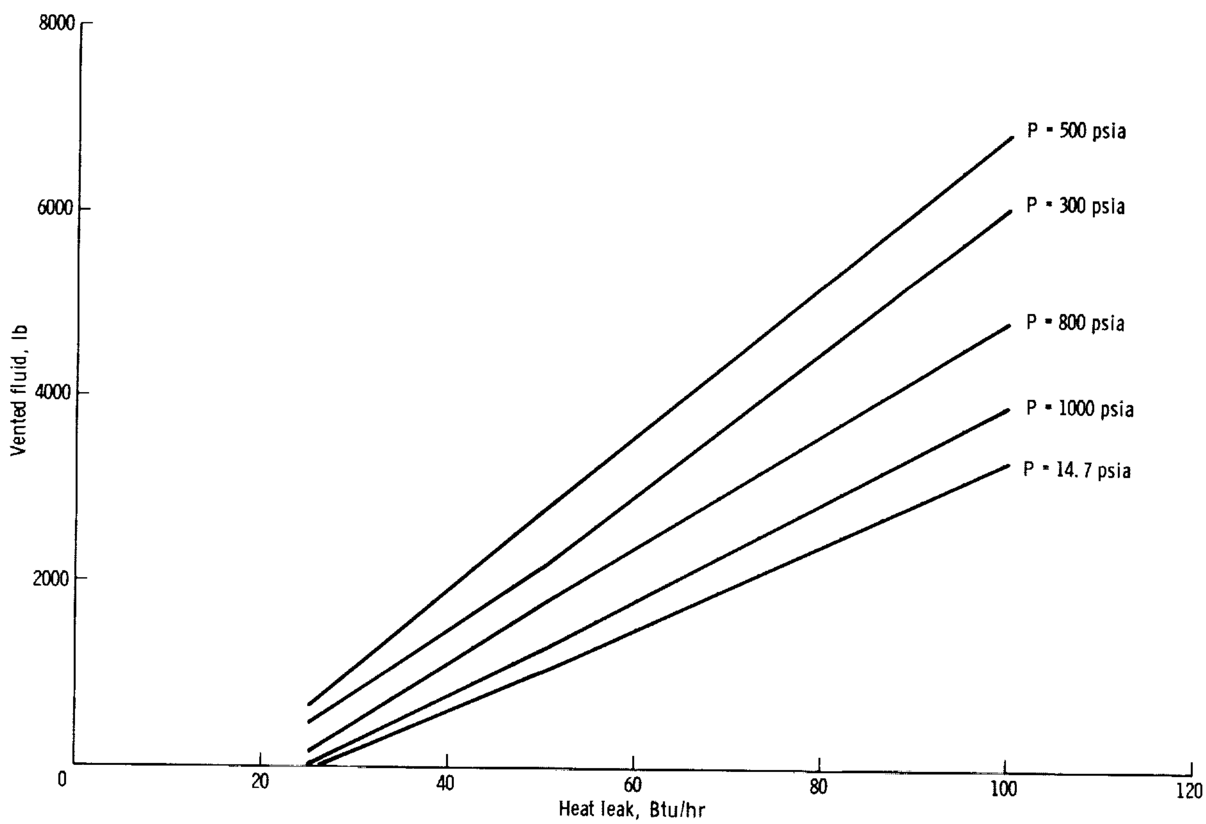


FIGURE 60.—Nitrogen flow characteristics for DRM IA.

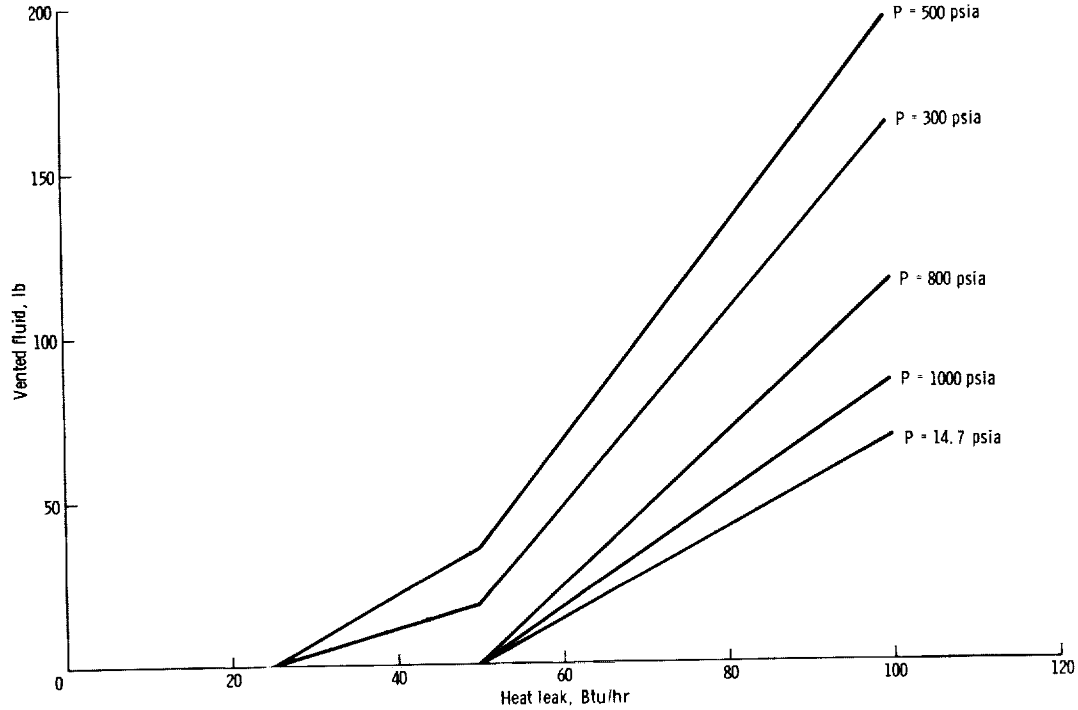


FIGURE 61.—Nitrogen flow characteristics for DRM IB.

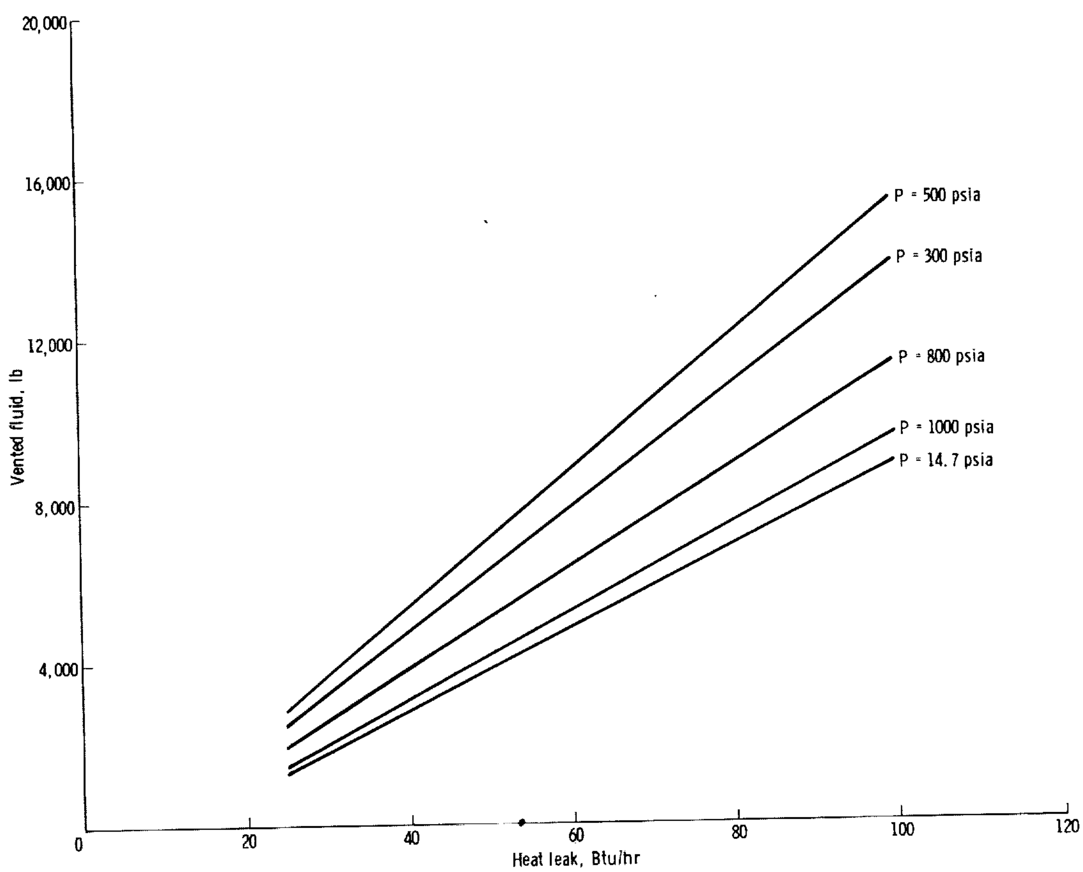


FIGURE 62.—Nitrogen flow characteristics for DRM IIA.

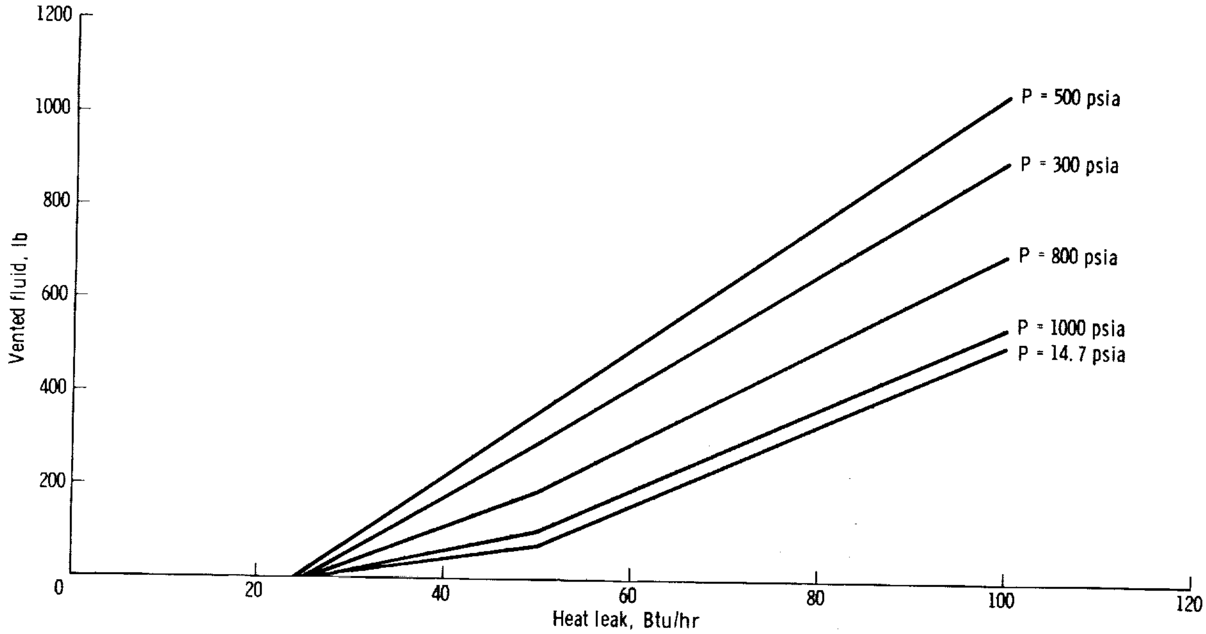


FIGURE 63.—Nitrogen flow characteristics for DRM IIB.

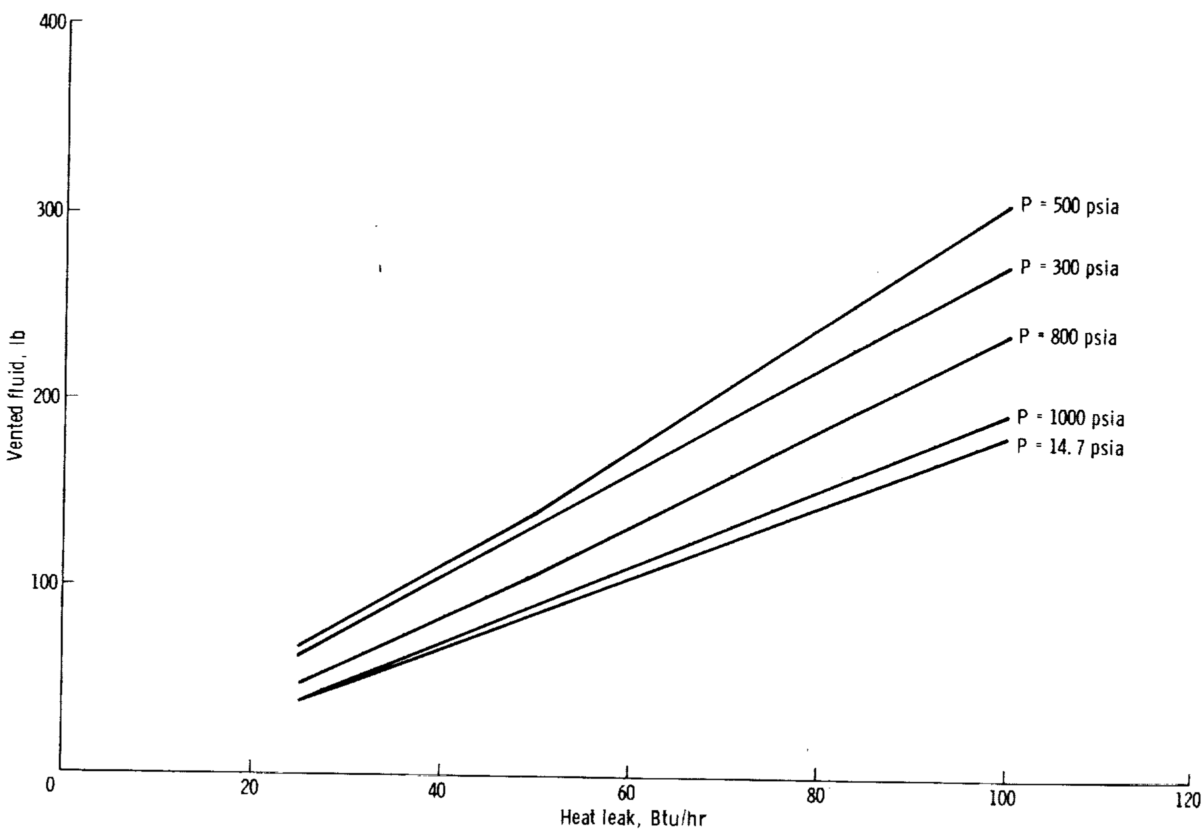


FIGURE 64.—Nitrogen flow characteristics for DRM IIC.

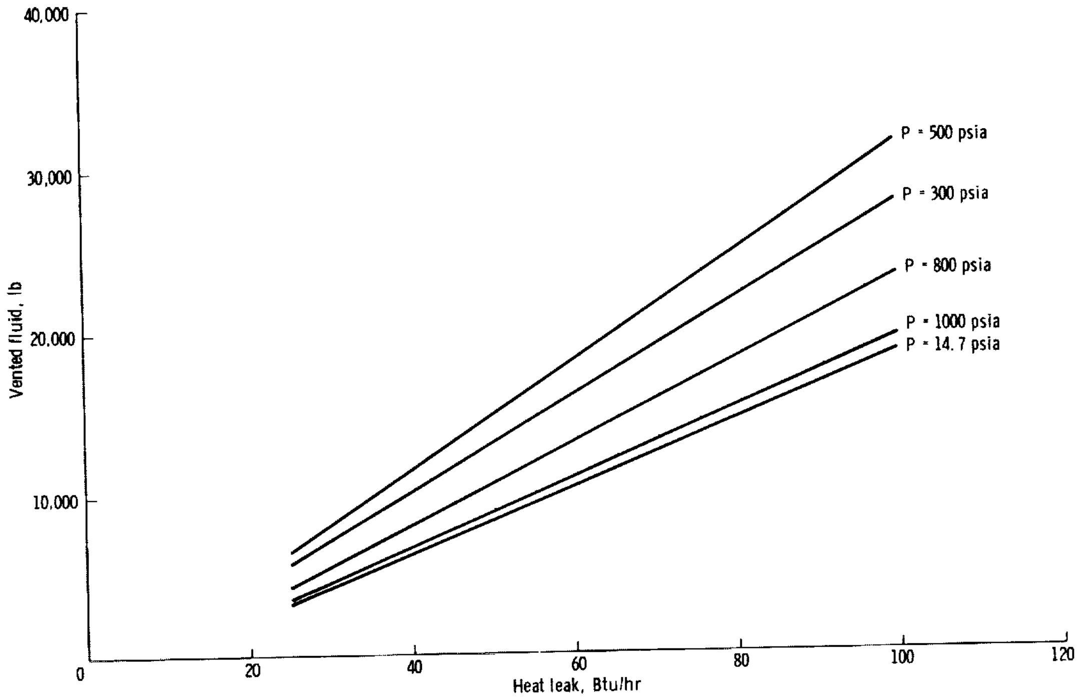


FIGURE 65.—Nitrogen flow characteristics for DRM IIIA.

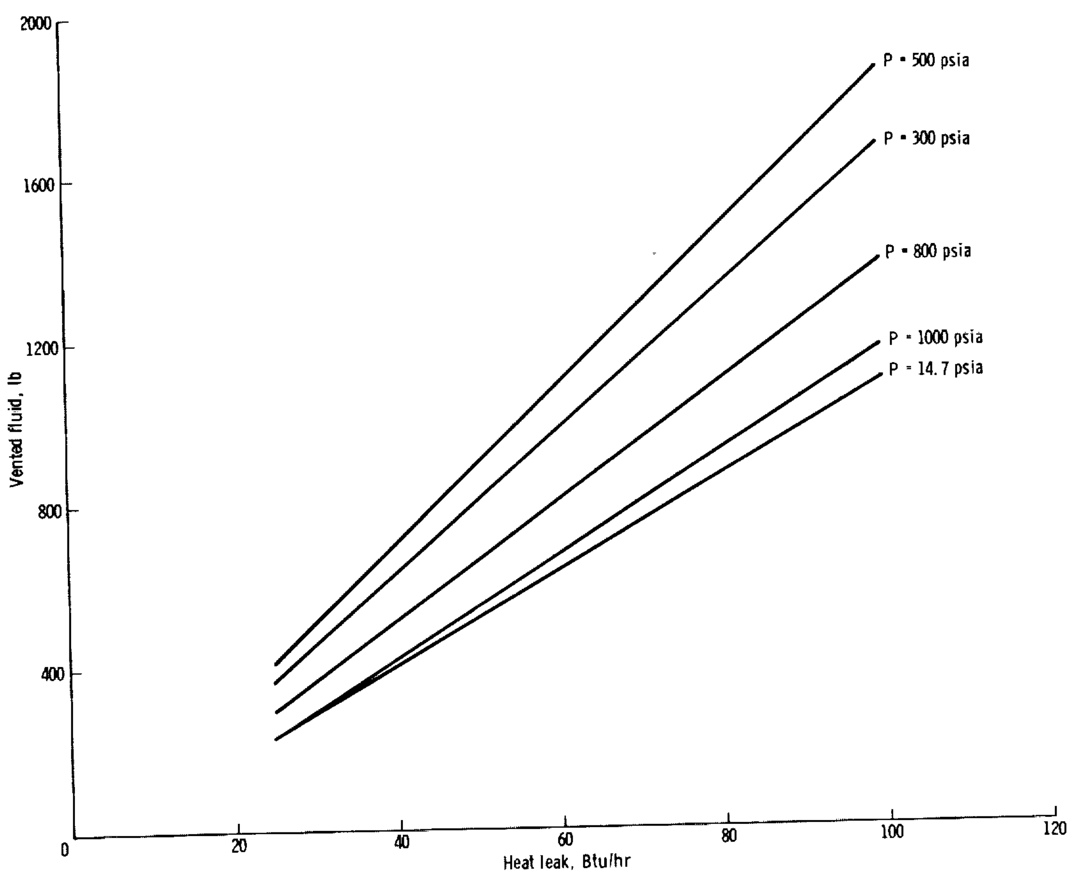


FIGURE 66.—Nitrogen flow characteristics for DRM IIIB.

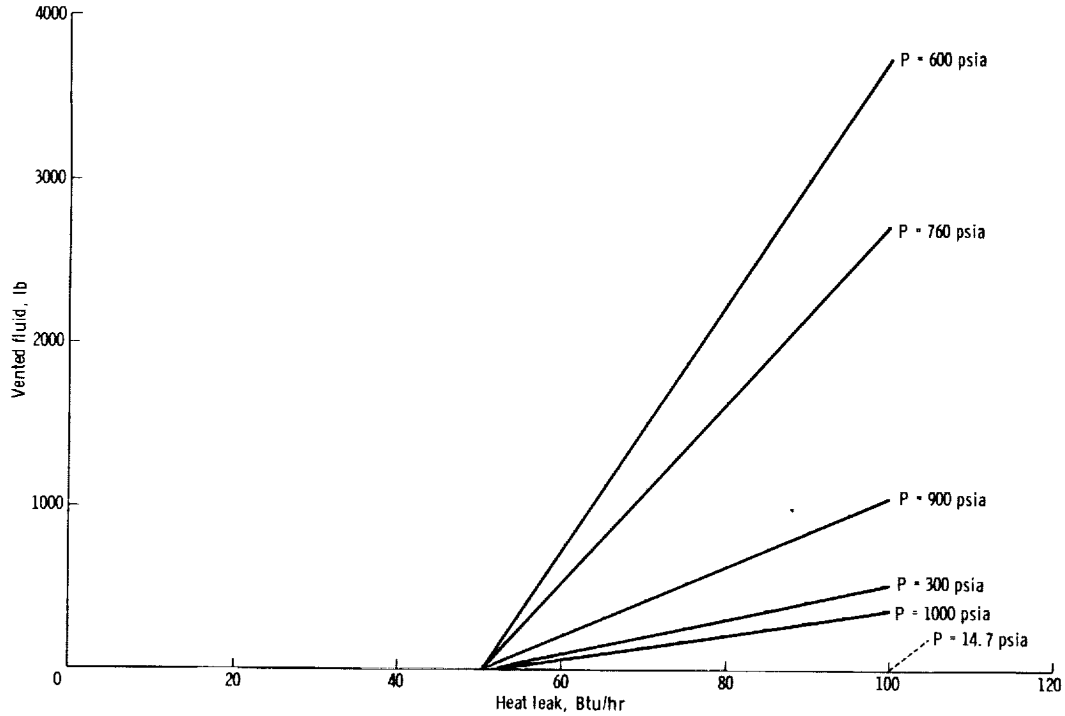


FIGURE 67.—Oxygen flow characteristics for DRM IIA.

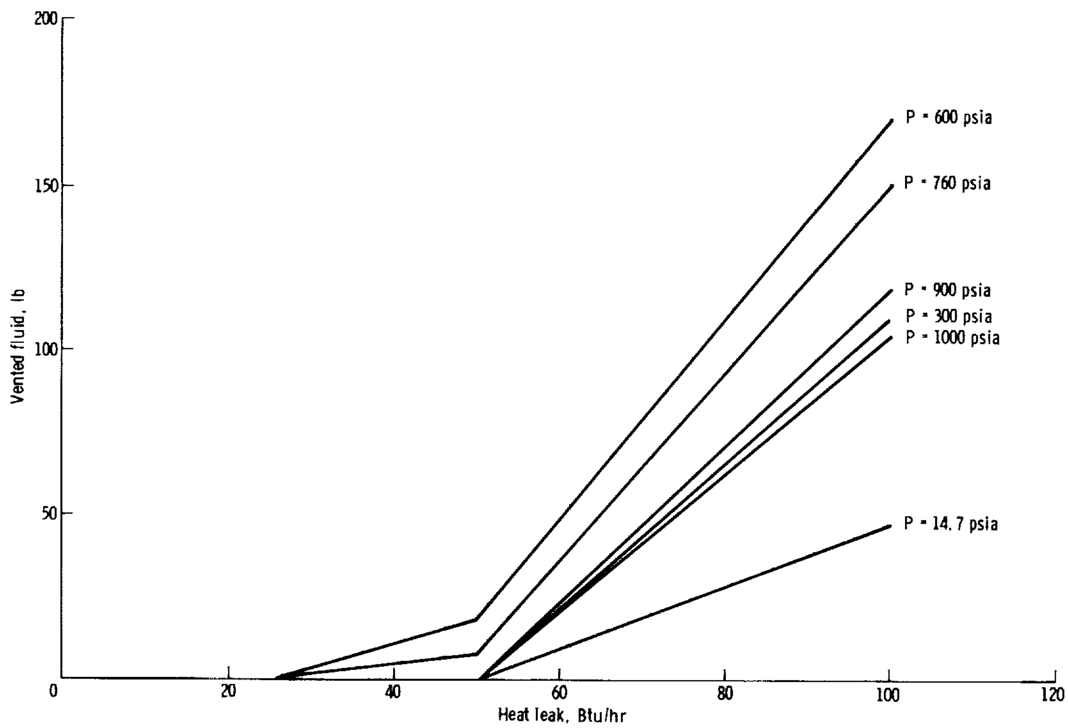


FIGURE 68.—Oxygen flow characteristics for DRM IIC.

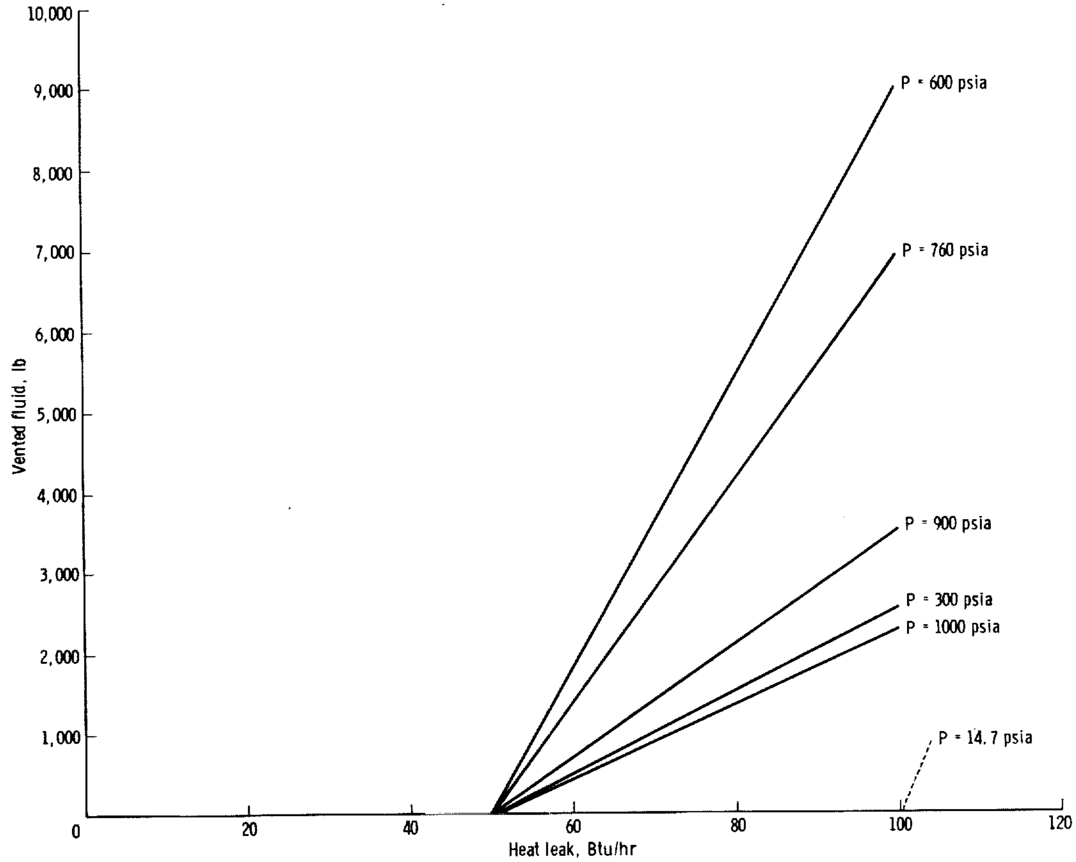


FIGURE 69.—Oxygen flow characteristics for DRM IIIA.

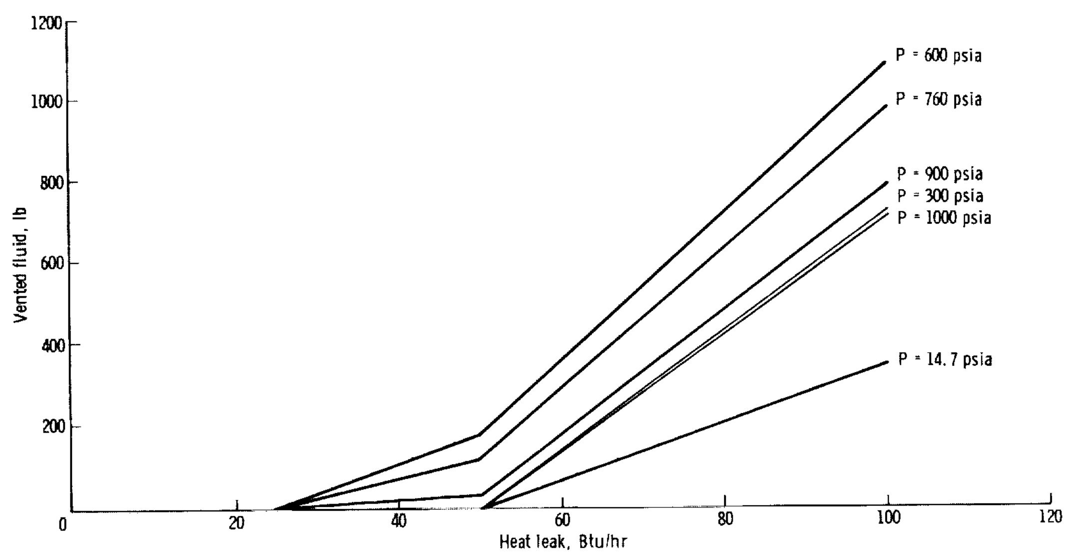


FIGURE 70.—Oxygen flow characteristics for DRM IIIB.

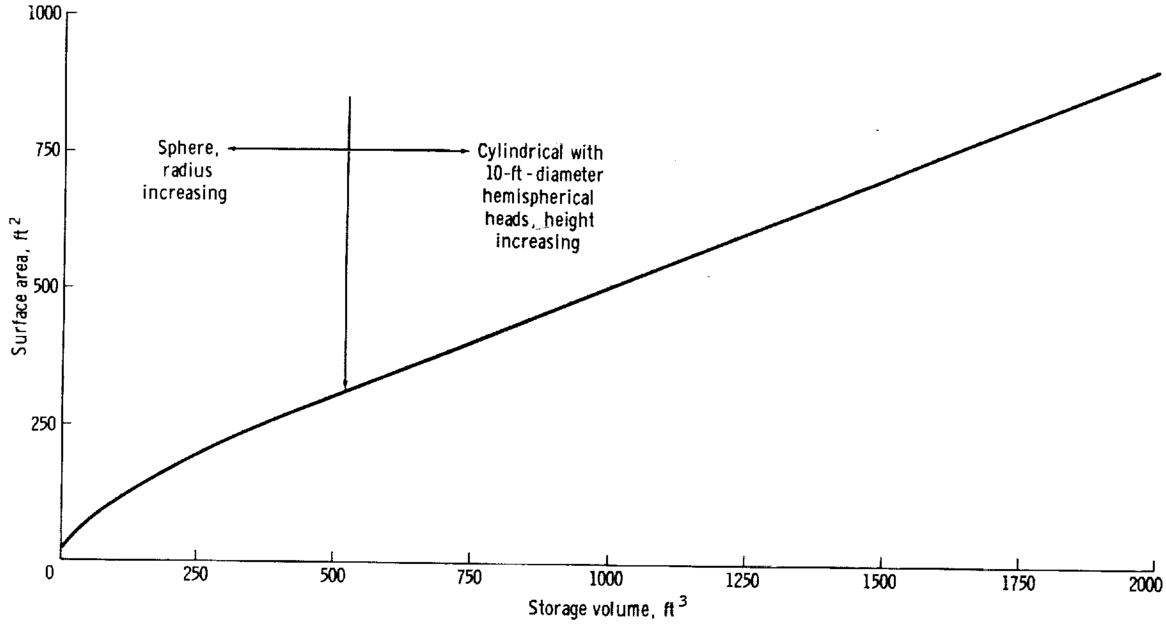


FIGURE 71.—Storage-vessel characteristics: surface area plotted as a function of storage volume.

may be exposed to thermal radiation. The total heat leak to the dewar divided by the surface area of the dewar is an excellent ratio to use when comparing storage-system performance and requirements. As the storage-vessel size is increased, the ratio Q_L/A ($Q_L =$ heat leak and $A =$ area) represents the design improvements that must be made to maintain the same level of performance. Values of these ratios are given in table 28 for the situations that are being discussed. A record of these ratios for vacuum-jacketed vessels that have been used on board previous spacecraft is shown in figure 72. This figure is included for comparison with the Q_L/A data that are given in the last row of table 28. As shown by this comparison, some major insulation advances or the addition of active refrigeration will be required for the long-duration design reference missions. Other design reference missions have relatively stringent requirements, but in the case of the Earth-to-orbit shuttle (DRM IB), the weight constraints are quite severe. As weight becomes critical, the possibility of using the conventional vacuum jacket wanes. In turn, the lack of the vacuum jacket raises the obtainable Q_L/A value. The results of the first-look study for the metabolic and possible hybrid-fuel-cell CGSS are shown in the design summary table (table 28), which includes the final thermal-protection and fluid requirements for the subject design reference missions.

Pressure-vessel weight.—In previous sections, several figures were presented that may be of use to the designer in the anticipation of the required thermal performance of a particular CGSS. The last step in the selection of storage pressures and the subsequent determination of thermal requirements involves consideration of the pressure-vessel weights. A weight comparison for 10-foot spheres and 10-foot-diameter cylindrical vessels that have overall lengths as long as 20 feet is shown in figures 73 and 74. Nominal values for typical pressure-vessel materials such as Ti–5Al–2.5Sn and 2219 aluminum were used. However, no attempt has been made to ascertain the weights of the thermal-protection components (insulation, vacuum shroud, shadow shield, special surfaces, and so forth). This weight value is related closely to the CGSS detail design, and, beyond prop-

TABLE 28.—*Design Summary—Thermal-Protection and Fluid Requirements*

Parameter	Design reference mission [a]						
	IA	IB	IIA	IIB	IIC	IIIA	IIIB
Minimum flow rate for metabolic requirements, lb/hr							
O_2	1.88	3.11	1.39	2.25	0.80	1.30	0.73
N_2	0.32	0.75	0.136	0.45	0.07	0.10	0.70
Design goal heat leak, Btu/hr							
O_2	100	100	[b]50 or 100	100	50	[b]50 or 100	[b]25 or 50
N_2	[b]25	[b]50	[b,c]10	25	(c)	[b,c]10	(c)
Design quantities, lb							
O_2	8124	[d]1022	11 995	1624	134	22 483	738
N_2	1394	126	1178	330	12	1783	71
Volume, including 5-percent ullage, ft^3							
O_2	120.1	[e]15.1	177.4	24.0	2.0	469.1	10.9
N_2	29.1	2.5	24.6	6.8	0.2	37.2	1.5
Heat-leak-to-vessel surface area ratio, Q_L/A, Btu/hr-ft^2							
O_2	0.8	6.60	0.6	4.2	25.3	0.2	4.6
N_2	0.9	20.1	0.4	3.7	(f)	0.3	(f)

[a] DRM IIIC is not applicable.
[b] At 14.7 psia
[c] Because the weight penalty that is associated with transporting diluent gas may be severe, the requirement may be deleted.
[d] Plus 55 lb H_2
[e] Plus 13 ft^3 H_2
[f] No requirement

erty data, no truly parametric assistance to the analyst is possible. Several excellent references for insulation weights and performance are given in the bibliography, and several technology programs that are devoted to thermal protection and instrumentation advances are given in the next section of part 2.

Supporting Research

Prior to the partial unification of supporting research for cryogenics technology for space exploration, insulation and materials improvements were carried out largely by universities and private industry. The Battelle Memorial Institute and the National Bureau of Standards are notable exceptions. However, Defense funds spent in private industry resulted in the first cryogenic systems for flight and provided the basis for the Mercury and Gemini generations of spacecraft. By 1963, the Apollo Program had contributed to a viable research program in support of storage and supply systems for liquid hydrogen and liquid oxygen. In addition to rapid increases in materials technology (as discussed in chapters 1 and 2), other advances were being made in the storage and supply of two-phase cryogens (figs. 53 to 58). Also, the important aspects of low-gravity and zero-gravity quantity gaging and pressure measurement were refined.

Today, as the age of relatively inexpensive Earth-orbital transportation systems proceeds into preliminary design, the following areas of supporting research are funded and

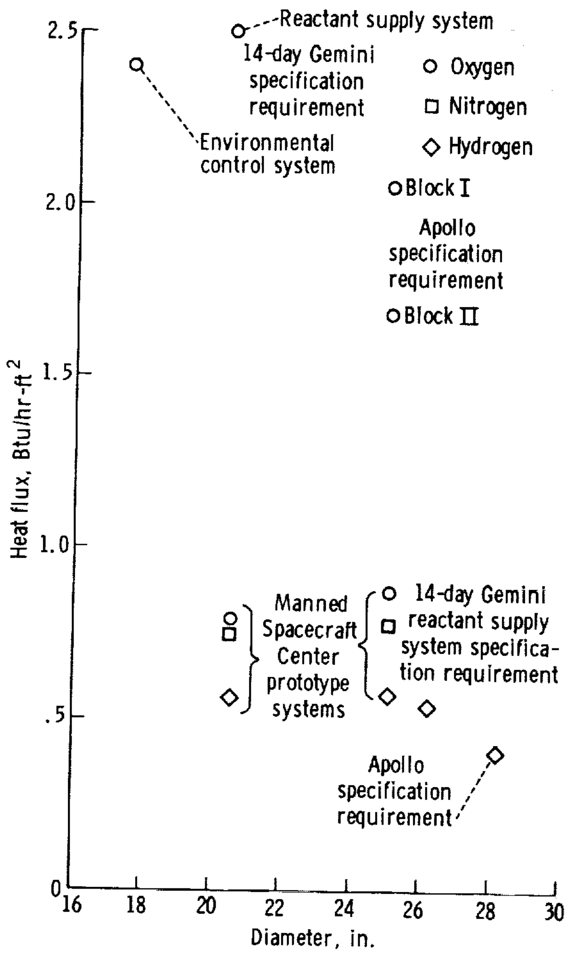


FIGURE 72.—Heat flux to the pressurized vessel plotted as a function of inner vessel diameter.

are administered by NASA engineering teams.

(1) Refinement of analytical techniques and materials for thermal insulation of the storage-system pressure vessel

(2) Long-life forced-convection devices

(3) Large-vessel manufacturing procedures and processes

(4) More efficient pressurization devices

(5) Retention and expulsion methods for two-phase (subcritical) storage of fluids in low gravity and zero gravity

(6) Quantity gaging for two-phase storage fluids in low gravity or zero gravity

(7) Storage-system refrigeration and reliquefaction of vented gases

In addition to this list, research into durability and reusability of sensitive and complex systems for uses such as the space shuttle has begun and encompasses the cryogenic-systems designs.

Summary

Part 2 of this chapter is a discussion and an analysis of the next phase of cryogenics technology for space flight. Because detailed plans are not available currently, the described missions that will be flown in the 1970's have been described parametrically when possible. Generally an attempt has been made in part 2 to show the direction that space exploration will take in the next decade, and, more specifically, what these tasks mean to cryogenics technology for flight. Accordingly, each design reference mission was shown to result in a set of flow and thermal-performance requirements. Several parameters were not discussed fully because they proliferate geometrically during the course of the study, and discussion of them does not serve the purpose of this portion of the report, which is to describe a systems-engineering approach to a first-look cryogenic-storage-system design and to describe near-future space-exploration goals and modes. Finally it has been shown that the systems-engineering approach results in data that can be compared with state-of-the-art cryogenic-systems data and that can be provided as input to the systems-engineering effort on the entire spacecraft. More advanced (far-future) missions and concepts cannot be discussed to the depth of analysis that has been shown herein. However, it is possible to define some limits, as will be seen in part 3 of this chapter.

PART 3—THE ADVANCED LUNAR BASE AND ADVANCED MANNED PLANETARY MISSIONS

In this section, the quantity and performance requirements of cryogenic gas storage systems are discussed in terms of missions that are still in the somewhat distant future. Specifically, emphasis is on metabolic oxygen (O_2) and the related mission-design concepts.

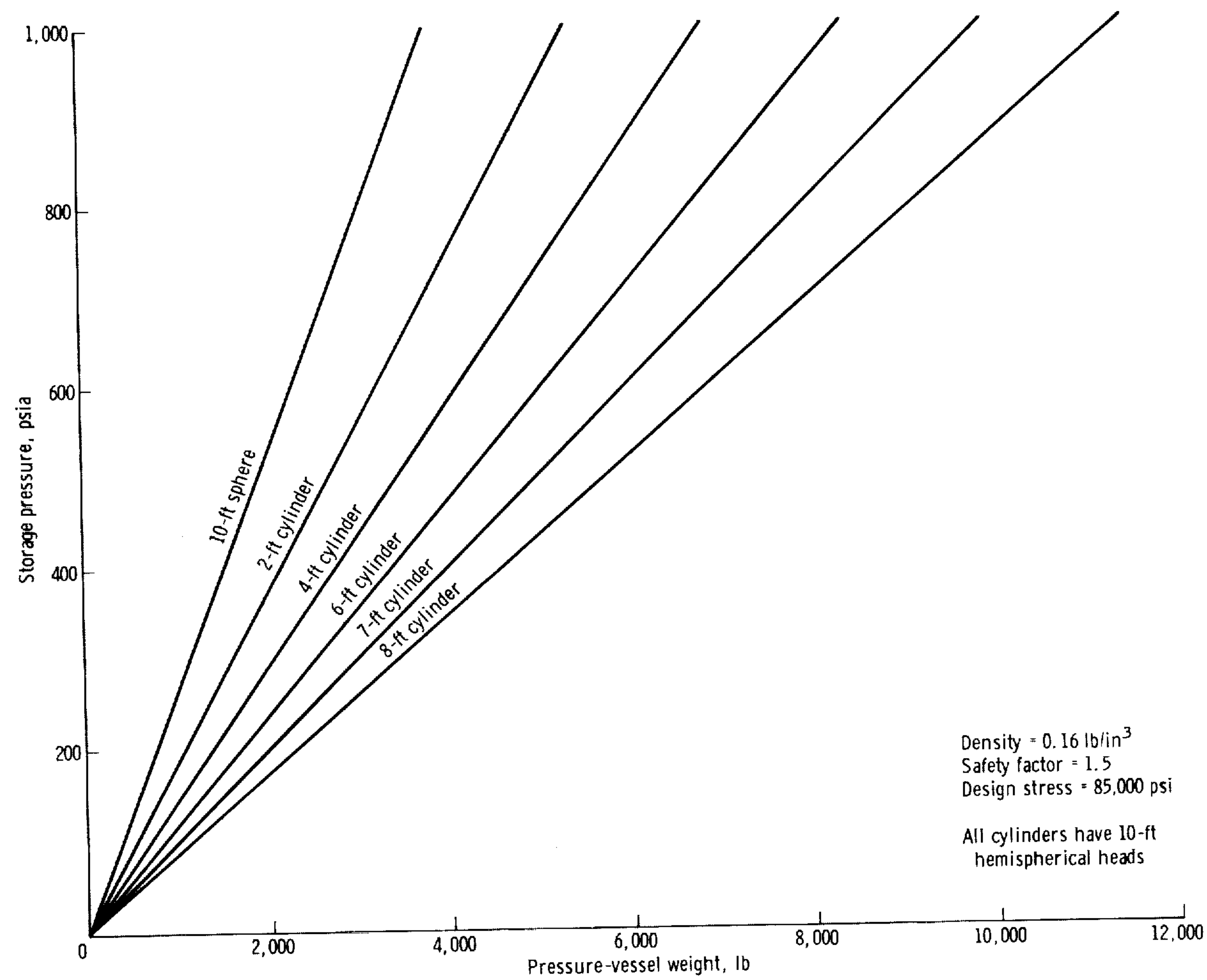


FIGURE 73.—Storage pressure plotted as a function of pressure-vessel weight for tanks made of Ti–5Al–2.5Sn.

The concepts presented for these missions are not to be construed as representing approved, funded objectives. Rather, this discussion is an extension of the mission-complexity aspects of near-future missions and early lunar missions that were discussed in part 2 of this chapter, and represents a primordial stage in the systems-engineering technique emphasized throughout this report. Quantification of engineering processes and physical phenomena is subordinated to the purpose of this part of this chapter, which is to discuss future possibilities for the use of cryogenic gas storage systems in space flight.

Cryogenic Gas Storage Systems for a Lunar Base

It has been proposed (ref. 1) that a crew of 24 scientists and engineers should have individual stay times that range from 3 to 24 months in a lunar base or farm. The crater Grimaldi has been proposed (ref. 1) as the best site for a manned lunar base. The location of the site does not impose a plane-change requirement, which makes it desirable from a trajectory viewpoint. Grimaldi is slightly south of the lunar equator, is close to the western limb, and is considered to be young relative to the age of the Moon. Thermal ex-

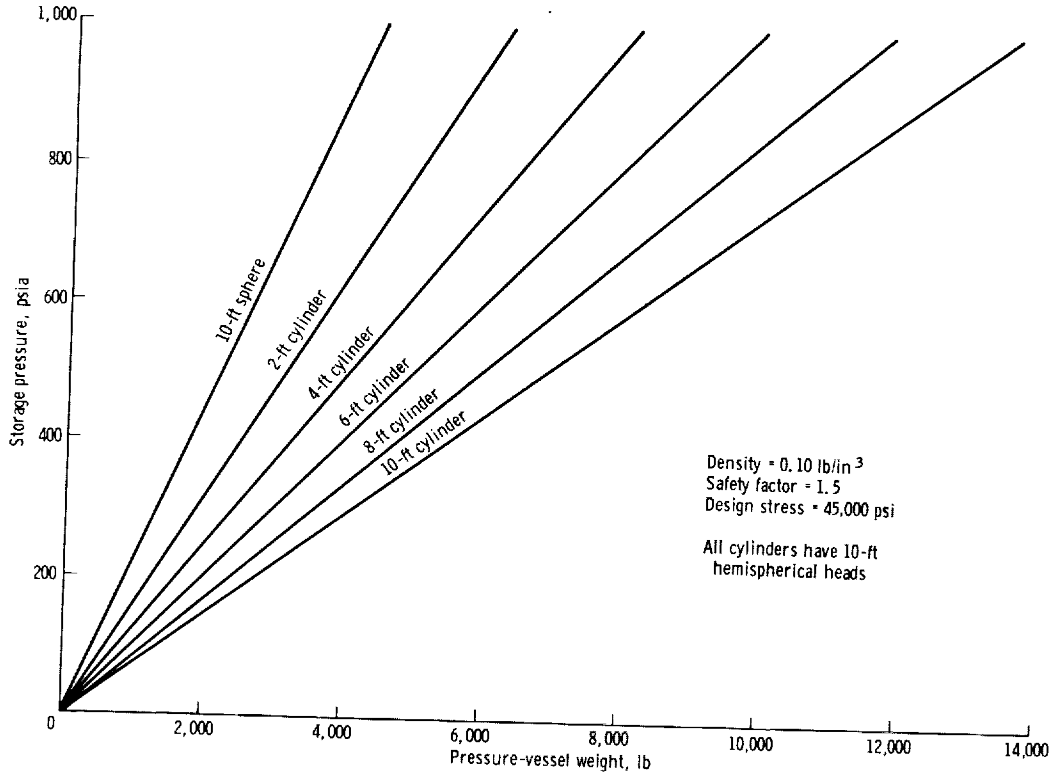


FIGURE 74.—Storage pressure plotted as a function of pressure-vessel weight for tanks made of 2219 aluminum.

tremes are characteristic of the site; however, the site is suitable from the astronomical viewpoint, and there are selenographic features of interest in the vicinity.

To the extent made possible by the installation of solar-energy converters and the development of a lunar farm, the self-contained functions should be centralized. However, the individual shelter units should be equipped with self-contained life-support and power-generation equipment to provide redundant capability. Cryogens such as liquid oxygen (LO_2) and liquid nitrogen (LN_2) are needed as standby atmosphere sources in the event of emergencies such as sudden decompression of the lunar-farm structures. As proposed in reference 1, the moonlab life-support system should be a closed system, but shelter leakage can be balanced by water and gas makeup. Carbon dioxide (CO_2) and oxygen balance can be established by the photosynthetic processes of crops grown on the lunar farm. Inflatable structures have been proposed for the growing areas, and gas exchange can be accomplished by circulation of the breathable atmosphere. Water of transpiration can be condensed in a heat exchanger; through this process, potable water can be collected and a significant heat load can be rejected. Diluted urine and processed fecal matter would be supplied to the soil of the farm. Thus, a closed system between the crewmen and the lunar farm can be established, as shown in figure 75 (which has been reprinted from reference 1 by permission).

A regenerative central power-supply system that has been proposed for the lunar base (ref. 1) is composed of hydrogen-oxygen fuel cells, solar cells, solar thermionic converters, and a water-electrolysis unit. Moonlab operations and the electrolysis unit could be powered by the conversion of solar energy during the 336-hour lunar day. Hydrogen and oxygen electrolysis products could be lique-

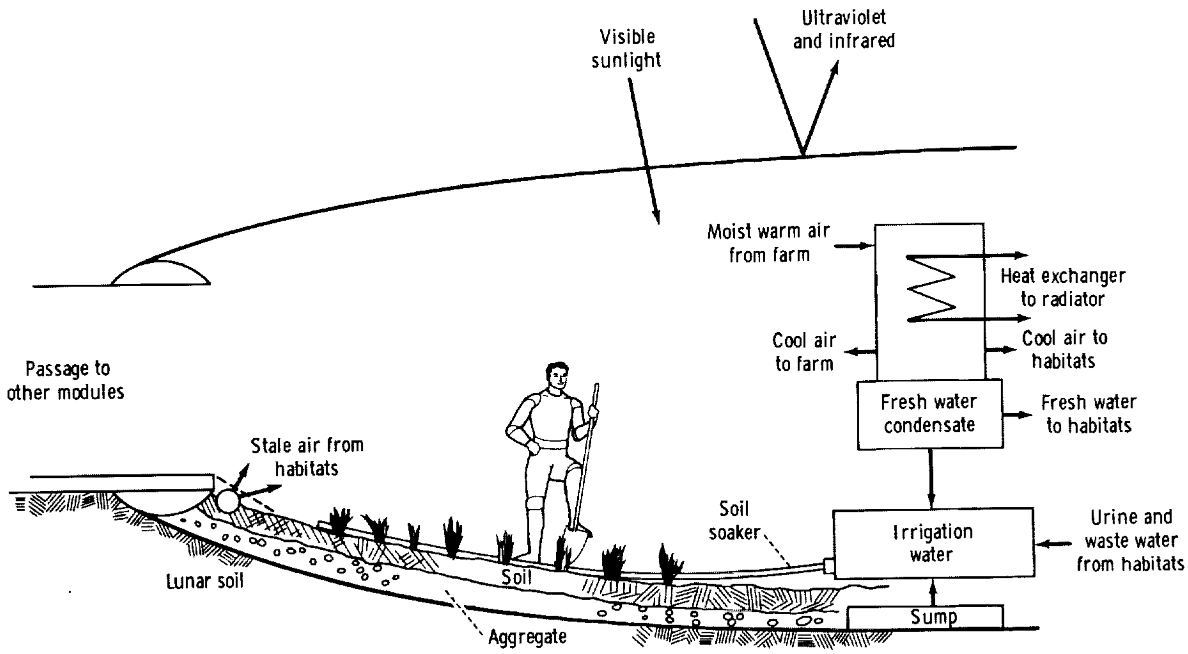


FIGURE 75.—Cross-sectional view of the lunar farm enclosure. Reprinted by permission of the copyright owner.

fied, stored cryogenically, and could be used, by the process of fuel-cell conversion, for power during the 336-hour lunar night. It has been noted (ref. 1) that the lunar-night powerload will determine the electrolytic and cryogenic-fluid-storage capacities. Because of the thermal extremes that are characteristic of Grimaldi, heating systems would consume more power than would cooling systems. Also, at night 1 kilowatt will necessitate 4 to 5 kilowatts of solar-energy-conversion capability; this situation is the result of the sum of the conversion inefficiencies of the component steps in a cascade process. The nighttime power needs, mainly heating load and emergency life-support-system requirements, would be in excess of 40 kilowatts. Direct solar-converter connection would result in a higher daytime power capacity (ref. 1).

One electrolytic cell that has been developed and could be used (ref. 2) can produce 3.4 kilograms of water per hour. The unit has the capability to electrolyze 1150 kilograms of water per 336-hour lunar cycle. To produce the predicted nighttime power minimum (40 kilowatts), 16.4 kilograms of reactants would be used per hour. For a 372-hour operating time, 6100 kilograms of fuel-cell reactants would be needed (ref. 1). The solar-cell array that has been designed (ref. 3) for use in conjunction with the electrolytic cell that has just been discussed must produce 33 kilowatts. Cryogenic tanks salvaged from the landing vehicle can be used for the storage of cryogens or electrolysis products.

If water is found in the lunar-surface material, crushing and roasting of the surface material could result in the extraction of the water. Electrolysis of this water could result in the production of hydrogen and oxygen gas. The power needed for operations may be supplied by the use of additional solar conversion units. An additional electrolytic workload would be imposed so that hydrogen and oxygen could be produced. Hydrogen and oxygen could be used in lunar-surface excursion vehicles. The reaction product (water) of the excursion vehicle cryogenic thrusters would not be recoverable; thus a closed system would not be feasible. Liquid hydrogen and liquid oxygen, derived from lunar-material-bound water, could be used to replenish propulsion, life-support, and fuel-cell systems. Thus a significant lift-off weight savings may be gained at the time of departure from Earth.

It is apparent that both solar and cryogenic power-generation methods will be needed at a manned lunar base. Cryogens must be maintained for emergency life-support purposes; therefore, the lunar CGSS should be designed to meet both the life-support and the power-generation requirements.

Lunar Storage of Cryogenic Fluids

It may be necessary to store cryogenic fluids for indefinite time periods on a nonterrestrial body. Because of the requisite storage equipment and because of the indeterminate nature of some of these nonterrestrial surfaces, this aspect of far-future missions will be considered only briefly. Although Mars, Mercury, and certain moons of Jupiter and Saturn may be involved ultimately in this problem, this discussion will be confined to problems that involve the storage of cryogenic fluids on the lunar surface. The extension of the following concepts of lunar storage of cryogenic fluids to use in the manned lunar laboratory and farm is of the utmost practicality.

To simplify the logistics vehicle problem storage tanks destined for use on a nonterrestrial body should be equipped with insulation that is suitable for thermal protection of cryogenic fluid while in space transit. On a nonterrestrial surface, several problems may occur: Conductive heat leak through the supportive structures, solar radiative heat transfer, and heat radiated from the surface itself are three such possible problems. Radiative heat transfer from the lunar surface would be a function of the selenographic position of the site. In lunar darkness, the selenographic position of the site would not be involved. The lunar surface can be covered with substances such as aluminized Mylar or titanium oxide powder to alter the local albedo.

The lunar-surface temperature has been calculated (ref. 4) for use in the solution of problems involving cryogen storage on a nonterrestrial surface. In these calculations, Ehricke assumed a flat lunar surface, neglected the inclination between the plane of the lunar equator and the plane of the ecliptic (approximately 1.5°), and assumed zero thermal conductivity for the lunar-surface material. Also, Ehricke noted (ref. 4): "In addition, on the surface of an airless body, it is relatively simple to alter the local thermal environment, whereas it is not as simple as it is in orbit to apply attitude control associated with a suitable tank shape." Ehricke proposed two fundamental methods for alteration of the local thermal environment: the use of shadow-casting bodies and the use of lunar-surface covers. Also Ehricke devised a formula for the calculation of the temperature of the lunar-surface cover. Ehricke concluded that the noon equatorial lunar-surface temperature would be a function of the absorptivity-to-emissivity ratio (α/ε) of the lunar surface. This conclusion is based upon the assumption that the lunar soil is at the same temperature as the lunar-surface-cover material. For comparison, several planetary reflectances (Bond albedos) are given in table 29. Obviously local surface temperatures affect heat leak to the stored cryogen and must be considered in addition to the albedos. For table 29, albedos for Mercury, Earth, Moon, and Saturn may be found in reference 5; for Mars in reference 6; for Venus in reference 7; for Jupiter in reference 8.

Lunar subsurface storage of cryogens.—It might be assumed that albedo and radiative-

TABLE 29.—*Bond Albedos*

Reflective Body	Bond albedo
Mercury	0.06
Earth	0.36
Moon	0.073
Mars	0.159
Venus	0.75
Jupiter	0.71
Saturn	Variable

heat-transfer problems might be avoided by the storage of cryogens in the lunar subsurface. However, subsurface storage of cryogenic fluids is not as effective as might be assumed. Neither low-boiling-point cryogens (such as helium or hydrogen) nor high-boiling-point cryogens (such as nitrogen) can be stored below the lunar surface without loss of thermal equilibrium. These fluids have equilibrium temperature values that differ significantly from the temperature of the lunar subsurface. Therefore, cryogenic fluids cannot be stored in the lunar subsurface for indefinite time periods. The time of storage-tank burial is significant because the temperature variation of the lunar subsurface material is dependent (in part) on the phase of the lunar cycle. A more detailed analysis of lunar storage of cryogenic fluids is given in reference 9, wherein several significant points are emphasized, and are summarized in the following two sections. Low-boiling-point cryogens and high-boiling-point cryogens involve two different problems regarding lunar-surface storage.

Low-boiling-point cryogenic fluids.—Relatively low-boiling-point cryogens (such as helium and hydrogen) have phase equilibrium temperatures that are lower than storage-tank-surface temperatures regardless of the lunar latitude at which the tanks are stored. Therefore, net heat flux is positive for all phases of the lunar cycle. Ideally, solar radiation should be reflected maximally and infrared radiation should be emitted maximally by the storage-tank surface; that is, the value of α/ϵ should be minimal. Also, the ideal tank shape should be such that minimal solar radiation (direct and albedo) is incident upon the tank surface. Titanium oxide powder would be a suitable lunar-surface covering because of its low α/ϵ ratio (approximately 0.2) and because this powder is not degraded by ultraviolet radiation.

High-boiling-point cryogenic fluids.—For a lunar cycle at a particular lunar latitude, the environmental temperatures and the equilibrium temperature of a given high-boiling-point cryogen (such as nitrogen) are within the same range of values. Therefore, if tank geometry, tank-surface characteristics, insulation, lunar latitude, and lunar-surface characteristics are chosen carefully, these cryogenic fluids may be stored at a relatively constant temperature. For these fluids, the converse of the low-boiling-point cryogen storage conditions is applicable; that is, storage at low latitudes, tanks that have large geometric shape factors with respect to the Sun and the Moon, and the exposure of the tank and nearby lunar surface to solar radiation are desirable conditions. For systems that have high thermal inertia, exposure to solar radiation during the lunar day balances heat loss during the lunar night so that thermal equilibrium is maintained.

Propulsion System Selection and Cryogens

Many factors must be considered before the final choice of a CGSS is made. Of these factors, the selection of the propulsion system is of particularly great significance for deep-space missions. Therefore, propulsion system selection will be discussed before planetary conceptual reference missions are evaluated. The selection of the propulsion system is based upon an evaluation of factors such as required velocity, required specific impulse (I_{sp}), mission duration, mission goals, and so forth. Subsequent to selection of a propulsion system, the requirements for and constraints upon the choice of cryogenic gas storage systems can be evaluated. Ehricke (ref. 10) has compared several propulsion systems with respect to suitability for interplanetary missions; this comparison is shown in table 30, but a detailed analysis of propulsion-system-selection techniques is beyond the scope of this report. (The data in table 30 were reproduced from reference 10 by permission of the American Astronautical Society.) A general comparison of propulsion-system characteristics and astronautical missions is shown in figure 76 (reprinted from reference 11 by permission). Crew size is based mainly on mission goal, and the velocity and specific impulse requirements are predicated upon optimized trajectories.

After selection of the appropriate propulsion system for a particular mission, cryogenic gas storage designs may be selected and optimized. Cryogenic gas storage systems may be required in conjunction with the propulsion system, the power-generation system, and the environmental control system. Or, the mission goals and engine type may be such that a CGSS is not required for propulsion or for power generation. The selection of the propulsion system and the power-generation system for a particular mission is an extremely complex procedure, and would constitute a separate report in itself. For this reason, only the metabolic-oxygen requirements will be discussed. Cryogenic quantities required for propulsion or power generation must be added to the metabolic quantities. The metabolic-oxygen quantities for the various conceptual reference missions were derived by multiplying the oxygen use rate (0.083 lb O_2/man-hr) by 24 for the oxygen quantity required per man day. The man-day requirement multiplied by the number of crewmen and the mission duration (in days) results in the overall metabolic-oxygen requirements for a particular CRM.

Planetary Conceptual Reference Missions

Although missions to all the planets have not been contemplated in detail, feasibility estimates can be made within reasonable limits and a CRM can be devised for a particular planet. Several possible conceptual reference missions will be described for each planetary mission under consideration. The conceptual reference missions that are discussed in this report have been arranged in three categories: single-planet missions, multiple-planet missions, and missions that involve a planetary satellite as the target. Not all conceptual reference missions involve a landing on the surface of the target planet; however, all missions have been grouped with respect to the target.

Mercury.—Manned missions to Mercury have been discussed elsewhere (refs. 10 and 12). One suggested mission (CRM IC) involves the establishment of a solar physics research station at the north pole of the planet. The mission goal necessitates 12 crewmen staying on the planet for 180 days; the mission duration is 220 to 260 days, and 5259 to 6215 pounds of liquid oxygen must be present for metabolic purposes. Four single-planet conceptual reference missions are shown in table 31. Because of the proximity of the Sun, power generation can be accomplished by the use of solar panels; therefore, cryogenic fluids will not be needed for power generation. Cryogenic gas storage systems will be needed for life support, however. For convenience, the following references may be consulted for more information on missions to Mercury: reference 12 for CRM IA, IB, ID, IIC, IIIC, IIID, IIIE, IVA, IVB, VA, and VB; reference 10 for CRM IC and IIA; refer-

TABLE 30.—*A Comparison of Propulsion-System Characteristics*

Engine type	I_{sp}, sec	Thrust acceleration, g	Propellant
Solid-core reactor (graphite or water-moderated)	750 to 850	$\gtrsim 1$	Hydrogen
Solid-core reactor (metal or metal carbide; nonmoderated)	850 to 1000	$\gtrsim 1$	Hydrogen
Fluidized-bed reactor	1000 to 1100	0.1 to 1	Hydrogen
Liquid-core reactor	1100 to 1200	10^{-4} to 10^{-3}	Hydrogen
Gaseous-core reactors	1700 to 2500	$\gtrsim 1$	Hydrogen
Nuclear pulse			
Fission pulse units	2500 to >5000	$\gtrsim 1$	Metal
Fusion pulse units	5000 to >10 000	$\gtrsim 1$	Metal
Controlled thermonuclear reactor	10 000 to >100 000	10^{-3} to 10^{-4}	Deuterium Helium-3
Nuclear electrostatic	5000 to >30 000	$\approx 10^{-4}$	Cesium Mercury

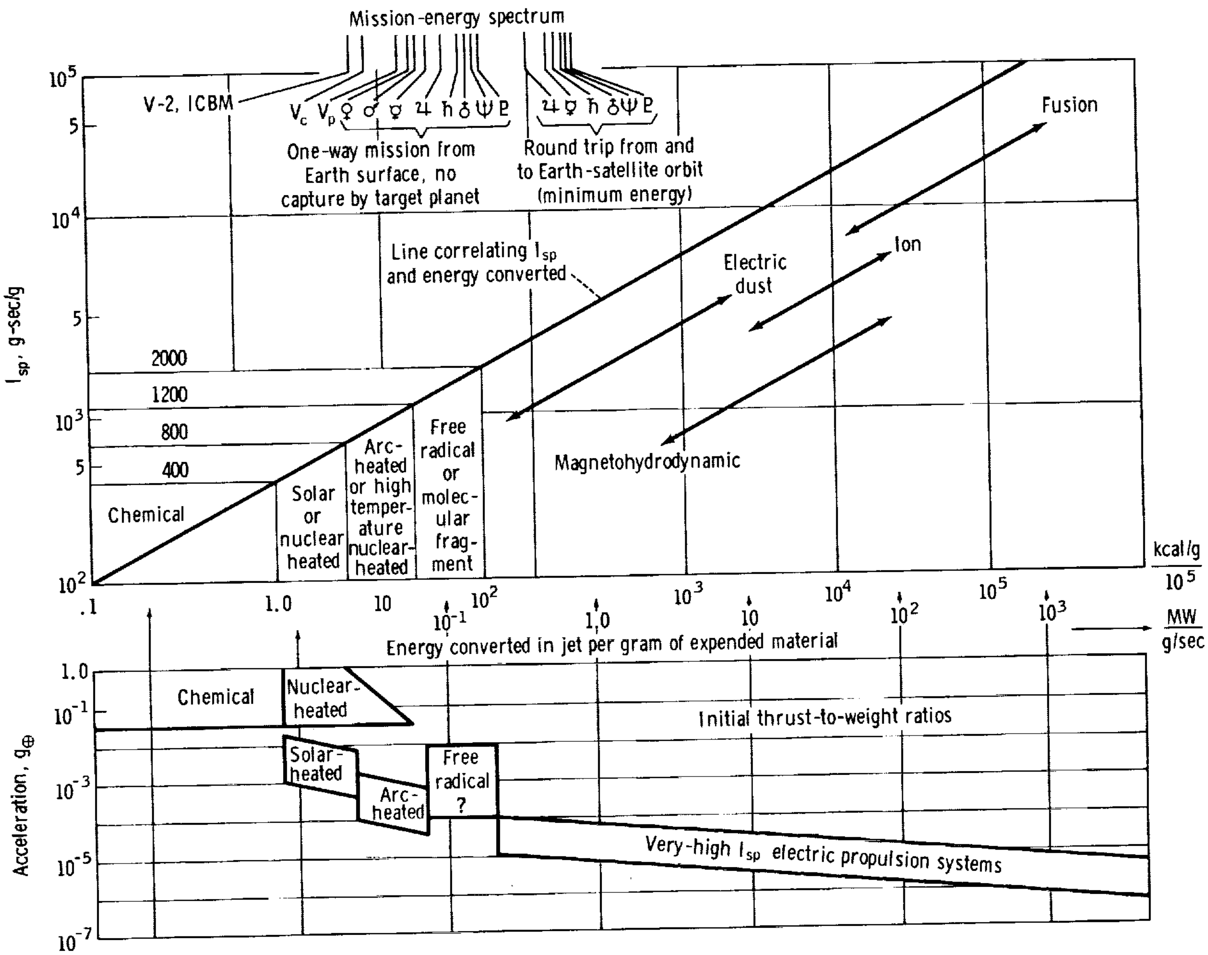


FIGURE 76.—Characteristics of propulsion systems and astronomical missions. ♀ = Venus, ♂ = Mars, ☿ = Mercury, ♃ = Jupiter, ♄ = Saturn, ♅ = Uranus, ♆ = Neptune, ♇ = Pluto, V = heliocentric velocity, c = circular, and p = parabolic. Reprinted by permission of the copyright owner.

ence 13 for CRM IIB and IIIA; and reference 14 for CRM IIIF. The metabolic-oxygen requirements for these missions are given in table 32, multiple-planet conceptual reference missions are summarized in table 33, conceptual reference missions VI and VII may be found in reference 13, and CRM VIII to XI may be found in reference 12. The metabolic-oxygen requirements for these missions are shown in table 34.

Venus.—Conceptual reference missions to Venus are designed for the purpose of studying the Venerean surface and environs (refs. 10, 12, and 13). A typical Venerean mission, CRM IIA, necessitates a crew of three men and a mission duration of 360 to 400 days. For such a mission, 2151 to 2390 pounds of liquid oxygen would be needed for metabolic purposes. Single-planet CRM profiles are given in table 31, and the metabolic-oxygen requirements for these missions are shown in table 32. Power could be generated both by the use of a nuclear-pulse propulsion system and by cryogenic means. Multiple-planet missions that involve Venus are shown in table 33; the metabolic-gas requirements for these missions are shown in table 34. These Venerean biplanet missions are in conjunction with Mercury or Mars, as shown in table 33.

Mars.—The goal of one proposed (ref. 10)

TABLE 31.—*Single-Planet CRM Profiles*

Target	CRM	Mission goal	Number of crewmen	Target stay time, days	Mission duration, days [a]
Mercury	IA	Brief exploration from orbit	6 to 8	(b)	220 to 260
	IB	Surface exploration	8 to 10	5 to 10	220 to 260
	IC	Solar physics research base	12	180	220 to 260
	ID	Synodic base	20 to 30	100 to 120	320 to 360
Venus	IIA	Surface investigation	3	(b)	[c]360 to 400
	IIB		3	(b)	400
	IIC		6 to 8	(b)	360 to 400
Mars	IIIA	Astrobiological base	5	(b)	400
	IIIB		8	(b)	440
	IIIC	Base supply; shuttle to Earth	6 to 8	(b)	360 to 400
	IIID		30 to 40	350 to 450	700 to 1000
	IIIE	Surface exploration	100 to 200	(b)	>1 synodic period
	IIIF	Surface exploration	70	400	969
Jupiter	IVA	Brief exploration from orbit	15 to 30	(b)	700 to 1000
	IVB	Extensive exploration from orbit	15 to 20	(b)	500 to 1000
Saturn	VA	Brief exploration from orbit	15 to 30	(b)	1000 to 1500
	VB		20 to 50	(b)	1000 to 1500

[a] Mission durations include stay times.
[b] Data not available or not yet defined.
[c] Arbitrarily selected.

TABLE 32.—*Metabolic-Oxygen Requirements for Single-Planet CRM Profiles*

CRM	Number of crewmen	Metabolic-O_2 use rate, lb/crew-day [a]	Mission duration, days	Metabolic-O_2 requirement, lb/CRM
IA	6	11.95	220	2 629
			260	3 108
	8	15.94	220	3 506
			260	4 143
IB	8	15.94	220	3 506
			260	4 143
	10	19.92	220	4 382
			260	5 179
IC	12	23.90	220	5 259
			260	6 215
ID	20	39.84	320	12 749
			360	14 342
	30	59.76	320	19 123
			360	21 513
IIA	3	5.98	360	2 151
			400	2 390
IIB	3	5.98	400	2 390
IIC	6	11.95	360	4 303
			400	4 781
	8	15.94	360	5 737
			400	6 374

TABLE 32.—*Metabolic-Oxygen Requirements for Single-Planet CRM Profiles—Concluded*

CRM	Number of crewmen	Metabolic-O_2 use rate, lb/crew-day [a]	Mission duration, days	Metabolic-O_2 requirement, lb/CRM
IIIA	5	9.96	400	3 984
IIIB	8	15.94	440	7 012
IIIC	6	11.95	360	4 303
			400	4 781
		15.94	360	5 737
			400	6 374
IIID	30	59.76	700	41 832
			1000	59 760
	40	79.68	700	55 776
			1000	79 680
IIIE	100	199.20	(b)	(b)
	200	398.40	(b)	(b)
IIIF	70	139.44	969	135 118
IVA	15	29.88	700	20 916
			1000	29 880
	30	79.68	700	55 776
			1000	79 680
IVB	15	29.88	500	14 940
			1000	29 880
	30	39.84	500	19 920
			1000	39 840
VA	15	29.88	1000	29 880
			1500	44 820
	30	79.68	1000	79 680
			1500	119 520
VB	20	39.84	1000	39 840
			1500	59 760
	50	99.60	1000	99 600
			1500	149 400

[a] Crew-day metabolic-O_2 use rate is based on an individual use rate of 0.083 lb O_2/man-hr.

[b] Data not available.

TABLE 33.—*Multiple-Planet CRM Profiles*

Target	CRM	Mission goal	Number of crewmen	Target stay time, days	Mission duration, days
Mars/Venus	VI	Exploration	5	—	590
Venus/Mars	VII	Exploration	3	—	590
Earth to Venus to Mercury to Earth	VIII	Exploration	8 to 10	(a)	350 to 450
Earth to Mercury to Venus to Earth	IX	Exploration	8 to 10	(a)	350 to 450
Earth to Mars to Venus to Earth	X	Exploration	8 to 10	(a)	500 to 650
Earth to Venus to Mars to Earth	XI	Exploration	8 to 10	(a)	500 to 650

[a] Stay time will be variable at Mercury or Mars, and will be zero for Venus.

TABLE 34.—*Metabolic-Oxygen Requirements for Multiple-Planet CRM Profiles*

CRM	Number of crewmen	Metabolic-O_2 use rate, lb/crew-day [a]	Mission duration, days	Metabolic-O_2 requirement, lb/CRM
VI	5	9.96	590	5 876
VII	3	5.98	590	3 526
VIII	8	15.94	350	5 578
			450	7 171
	10	19.92	350	6 972
			450	8 964
IX	8	15.94	350	5 578
			450	7 171
	10	19.92	350	6 972
			450	8 964
X	8	15.94	500	7 968
			650	10 358
	10	19.92	500	9 960
			650	12 948
XI	8	15.94	500	7 968
			650	10 358
	10	19.92	500	9 960
			650	12 948

[a] Crew-day metabolic-O_2 use rate is based on an individual use rate of 0.083 lb O_2/man-hr.

Mars mission (CRM IIIA) is the founding of an astrobiological research base. Five crewmen would spend 400 days on such a mission, and would need 3984 pounds of liquid oxygen for metabolic purposes. A mission for Martian-surface exploration (CRM IIIF) would involve 70 crewmen and a 400-day mission duration. For CRM IIIF, 135 118 pounds of liquid oxygen would be required for metabolic supply. Martian CRM profiles and the associated metabolic-oxygen requirements are given in tables 31 and 32. Multiple-planet conceptual reference missions that include Mars are summarized in tables 33 and 34. It has been stated (ref. 10) that a hybrid nuclear-pulse propulsion system/cryogenic power-generation method would be feasible for one of these missions. Power could be generated by nuclear means for the base, and metabolic gas could be supplied by cryogenic means. In reference 14, von Braun analyzed a Mars mission, aspects of which have been summarized in tables 31 and 32 (CRM IIIF). This mission analysis was published in 1962, and is predicated on the use of a hydrazine/nitric acid propulsion system. Such a system has a comparatively low specific impulse. However, von Braun considered this during mission design, and avoided the use of liquified-gas propellants so that collapsible propellant-storage tanks could be used. According to von Braun (ref. 14), the use of collapsible tanks would facilitate the construction of interplanetary vessels while in Earth orbit. Also, von Braun intentionally does not optimize parameters for this mission.

Jupiter.—Two Jovian conceptual reference missions and the appropriate metabolic-gas requirements are given in tables 31 and 32. These missions are orbital reconnaissance missions, not landing missions. Because of the distance (5.2 astronomical units) from the Sun (ref. 5), solar panels are not suitable for power generation. These missions involve nuclear-pulse propulsion systems (ref. 10). The more conventional propulsion systems do not achieve sufficient velocity for use on missions that involve distances as great as those that separate the terrestrial planets from the Jovian planets. Because of the distance from Jupiter to the Sun, solar panels are not of benefit.

Saturn.—Two Saturnian conceptual reference missions are outlined in table 31, and the metabolic-oxygen requirements are presented in table 32. As is the case for the Jovian missions, the Saturnian missions are for orbital reconnaissance. Solar panels are not feasible on Saturnian missions because of the distance from the Sun (9.6 astronomical units). Nuclear-pulse drive will be a good candidate propulsion system (ref. 10) because of the great distances that are involved in Saturnian missions.

Jupiter-Saturn convoy.—A convoy to Jupiter and Saturn has been suggested (ref. 10). The mission is to the fourth moon of Jupiter (Callisto) and to the sixth moon of Saturn (Titan). By the use of nuclear-pulse drive, the convoy would travel toward Jupiter and would separate at Jupiter approach. The Titan-bound vehicle would undergo a gravitational encounter with Jupiter while in peri-Jovian space. Ehricke (ref. 10) states that the Callisto-bound units "enter into an elliptical orbit capture mode and subsequently orbit-maneuver to the moon Callisto." The profile for this mission, CRM XV, is given in table 35. In table 35, some data for CRM XIIA, XIIB, XIIIC, and XIV were taken from reference 12; some data for CRM XIIIA and XIIIB were taken from reference 15; and some data for CRM XV were taken from reference 10. The metabolic-gas requirements, based on a crew size and a mission duration that were selected for purposes of illustration, are given in table 36.

Callisto.—The Jovian mission described by Ehricke (ref. 10) involves Callisto, the fourth moon of Jupiter, not Jupiter itself. Ehricke has described missions to Callisto and Hera (the seventh moon of Jupiter). The purpose of the Callisto mission is to study Jupiter and to establish a base for the exploration of the Jovian planets. The mission involves the use of a nuclear-pulse propulsion system.

The mean planetary surface temperature of Jupiter has been shown to be 105±3° K (ref. 8). For comparative purposes, 105° K $\approx -168.1°$ C $\approx -334.4°$ F. Ganymede (the third moon of Jupiter), Callisto, and Hera should be cool because of their distance from the Sun and because, if they have atmospheres, the atmospheres are likely to be tenuous. Therefore, CCGS installations that have minimal shielding are feasible on the surface of these Jovian satellites. If the Jovian manned station becomes a base for the exploration of the Jovian systems, these conditions will be fortuitous indeed. Large cryogenic-gas depots on the Jovian satellites will be feasible. For the exploration of other Jovian satellites, cryogenic-propellant pro-

TABLE 35.—*Planetary Satellite CRM Profiles*

Target [a]	CRM	Mission goal	Number of crewmen	Target stay time, days	Mission duration, days
Phobos	XIIA	Exploration	8 to 10	10 to 20	420 to 460
	XIIB	Exploration	8 to 10	10 to 20	800 to 1000
Deimos	XIIC	Exploration	5 to 10	8 to 10	420 to 460
	XIID	Exploration	5 to 10	8 to 10	800 to 1000
Ganymede	XIIIA	Base for Jovian-system exploration	5	(b)	800
Callisto	XIIIB	Base for Jovian-system exploration	8	(b)	800
Callisto	XIIIC	Exploration	20 to 30	30 to 60	500 to 1000
Titan	XIV	Exploration	50 to 70	60 to 80	1000 to 1500
Callisto/Titan	XV	Convoy and exploration	25/25	Variable	1500

[a] Phobos=Mars I, Deimos=Mars II, Ganymede=Jupiter III, Callisto=Jupiter IV, and Titan=Saturn VI.
[b] Questionable stay time.

TABLE 36.—*Metabolic-Oxygen Requirements for Planetary Satellite CRM Profiles*

CRM	Number of crewmen	Metabolic-O_2 use rate, lb/crew-day [a]	Mission duration, days	Metabolic-O_2 requirement, lb/CRM
XIIA	8	15.94	420	6 693
			460	7 331
	10	19.92	420	8 366
			460	9 163
XIIB	8	15.94	800	12 749
			1000	15 936
	10	19.92	800	15 936
			1000	19 920
XIIC	5	9.96	420	4 183
			460	4 582
	10	19.92	420	8 366
			460	9 163
XIID	5	9.96	800	7 968
			1000	9 960
	10	19.92	800	15 936
			1000	19 920
XIIIA	5	9.96	800	7 968
XIIIB	8	15.94	800	12 749
XIIIC	20	39.84	500	19 920
			1000	39 840
	30	59.76	500	29 880
			1000	59 760
XIV	50	99.60	1000	99 600
			1500	149 400
	70	139.44	1000	139 440
			1500	209 160
XV	50	99.60	2000	199 200

[a] Crew-day metabolic-O_2 use rate is based on an individual use rate of 0.083 lb/man-hr.

pulsion systems may be the most efficient; for expeditions to the outer boundary of the Jovian group, a propulsion system such as nuclear pulse or controlled thermonuclear reactor (CTR) may be the system of choice.

Titan.—The mission is not to the primary (Saturn), but is to Titan, the fourth satellite of the planet. The mission goal will be to study Saturn; generally the mission is similar to the Jupiter IV (Callisto) mission (tables 35 and 36).

Required Improvements

The CRM metabolic-gas requirements must be discussed relative to overall CRM goals and requirements. This discussion will place in perspective the component factors that are involved in CRM design. The minimal metabolic-oxygen requirements for five, 10, and 25 crewmen on missions lasting as many as 1500 days are shown in figure 77. In figure 78, the minimum metabolic-oxygen requirements are given for 200-, 500-, and 750-day missions and a variable crew size.

Ehricke (ref. 12) has devised an equation that is descriptive of the crew-size change associated with an increase in mission period. This equation is useful in the computation of the trade-off between crew size and mission period for a particular set of CGSS capabilities. Fluid (LO_2) weight is shown as a function of storage in figure 79. Also in figure 79, the minimum metabolic-oxygen requirements are shown for each of the three CRM cate-

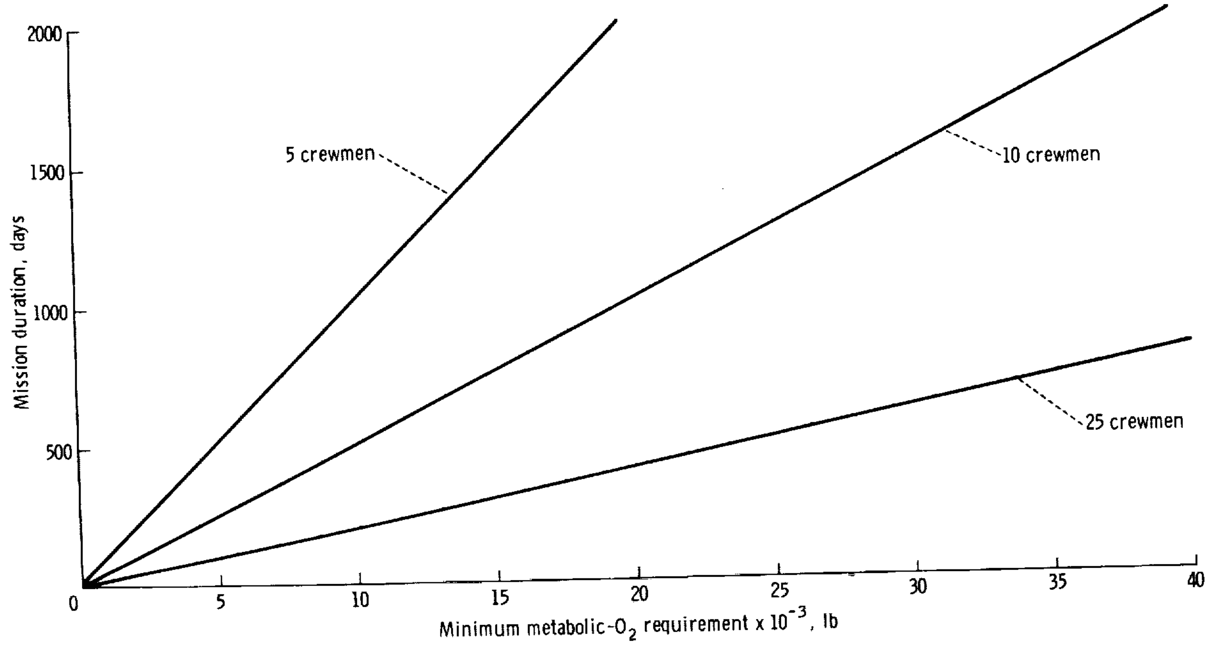


FIGURE 77.—Minimum metabolic-O_2 requirement for 5, 10, and 25 crewmen on missions of variable duration.

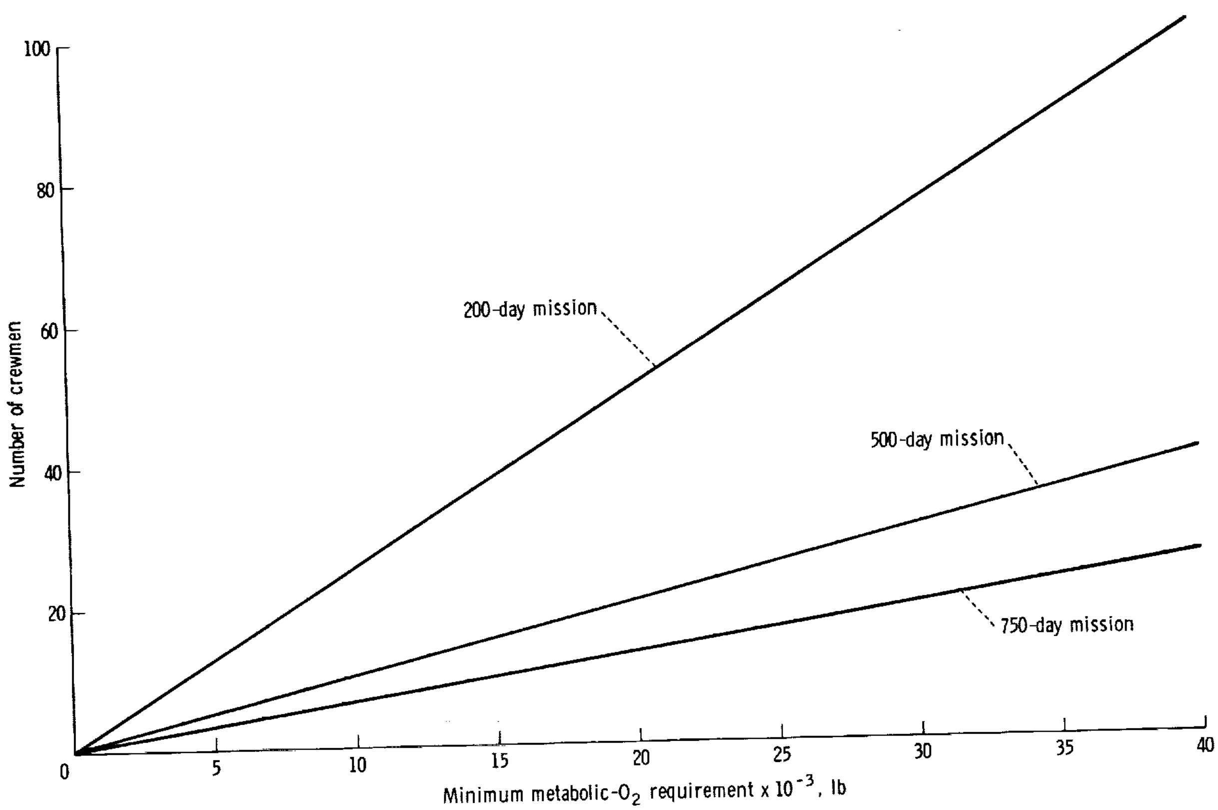


FIGURE 78.—Minimum metabolic-O_2 requirements for variable crew sizes on 200-, 500-, and 750-day missions.

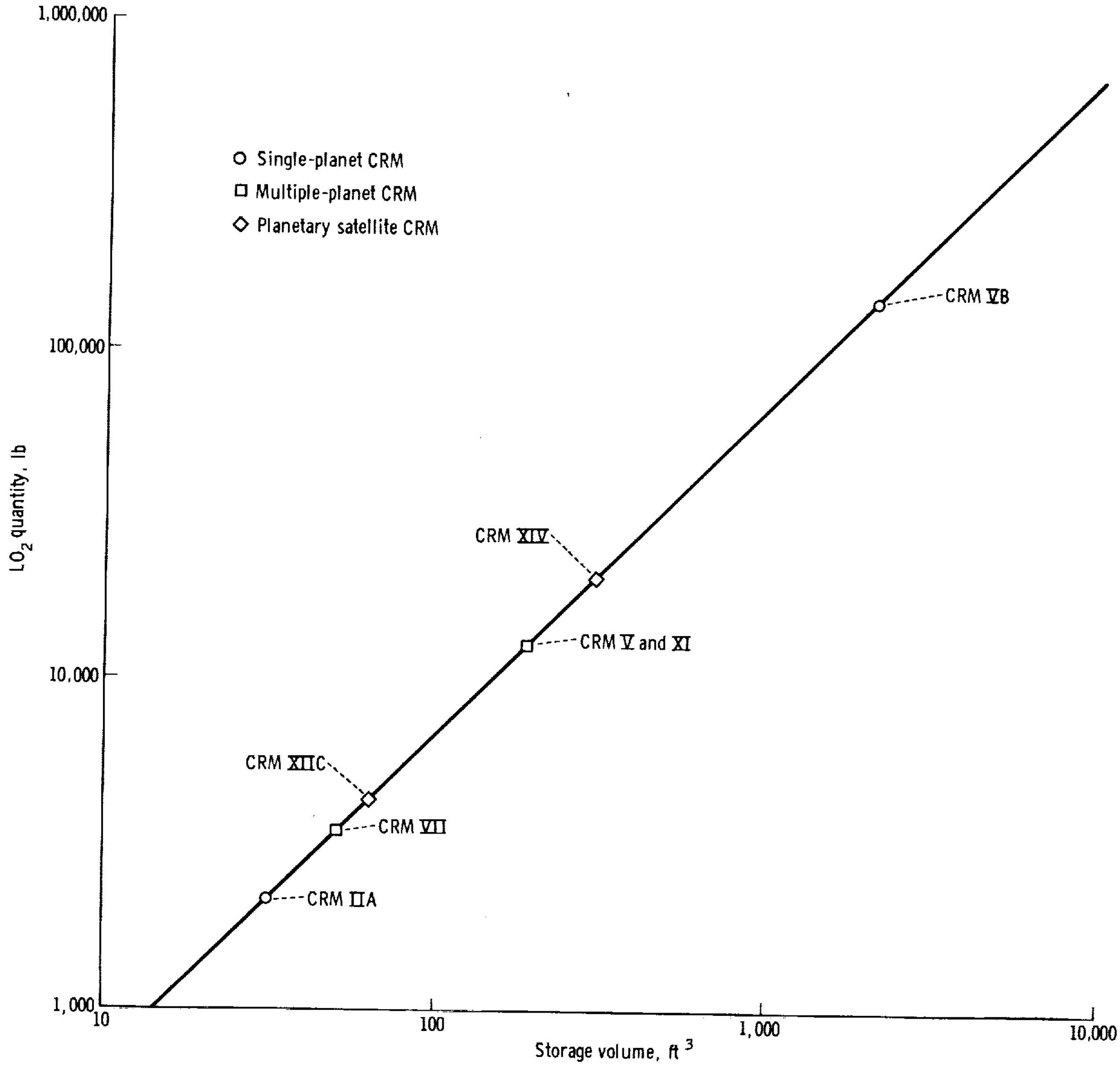


FIGURE 79.—Comparison of the LO_2 quantity and the storage volume that are required for the three CRM categories (based on 5-percent ullage).

gories and the minimum and maximum values within each CRM category are identified. Cryogenic or nuclear power-generation systems will be necessary, as will cryogenic sources for metabolic oxygen. Because of the distance from the Sun, solar panels are unsuitable for a Saturn mission. A summary of the optimized CGSS data for a mission to Titan is presented in tables 35 and 36.

Generally the value of heat leak per unit area $Q_L/A=0.20$ Btu/hr-ft² is representative of the current state of the cryogenics art. As an aid in the assessment of the existing capability to meet CRM duration requirements, the 100-percent-boiloff times for several cryogenic fluid (LO_2) quantities were calculated for $Q_L/A=0.20$ and 0.10 Btu/hr-ft². The results are shown in figure 80 (fluid quantity as a function of time). The maximum fluid quantity considered was 10 000 pounds. This quantity is small compared with the metabolic-oxygen requirements for conceptual

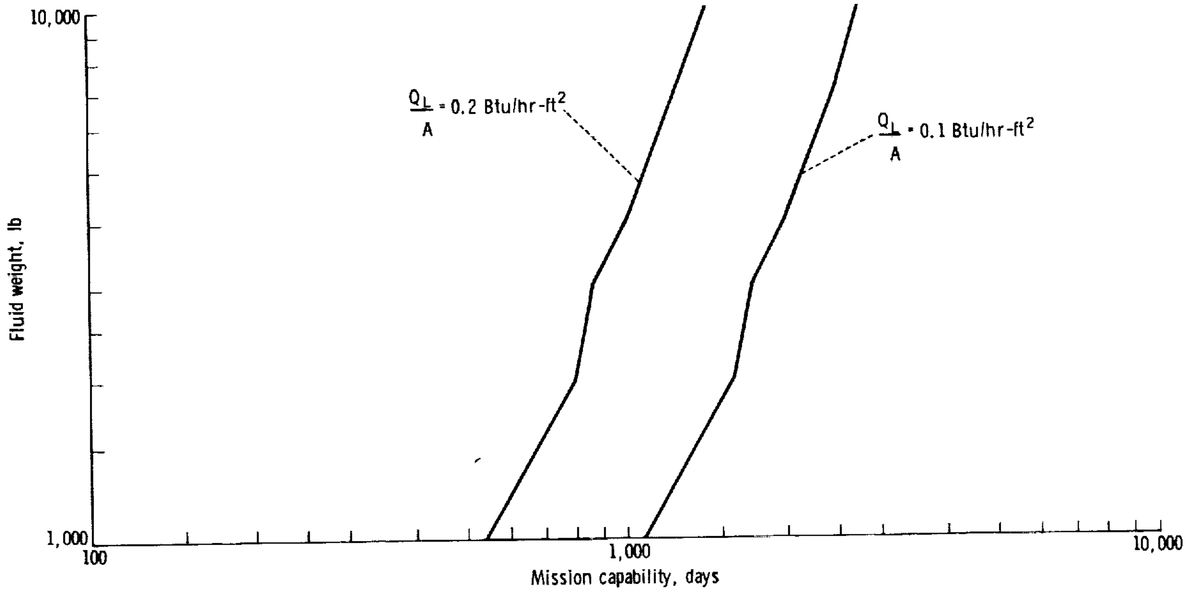


FIGURE 80.—Total LO_2 boiloff times for two heat-leak values.

reference missions such as IVA (Jupiter), VA (Saturn), VB (Saturn), XIIIC (Callisto), XIV (Titan), and XV (Callisto/Titan). In these calculations, extensive use was made of figure 71 (surface area as function of storage volume). In the generation of figure 71, the tank was assumed to be spherical and the diameter was increased to 10 feet; beyond this size, the tank was assumed to be cylindrical and to have 10-foot-diameter hemispherical heads. Also, the tank was assumed to be single walled. In the generation of figure 80, the fluid volume was calculated from selected fluid quantities and known fluid density. From this calculated fluid volume, figure 71 was used to determine the corresponding surface area of the storage tank. It should be noted that values of Q_L/A are not extremely accurate when extrapolated greatly for a wide range of tank sizes. The requisite metabolic-oxygen quantity for various conceptual reference missions may be located on the ordinate and the mission duration may be located on the abscissa of figure 80. If the intersection of the two points under consideration lies below (or to the right) of the Q_L/A=0.2 Btu/hr-ft² curve, the CRM under consideration is beyond the current state of the cryogenics art. If the intersection of the point lies above (or to the left) of this curve, the CRM metabolic-oxygen requirements lie within the current state of the cryogenics art. The points should be referenced with respect to the Q_L/A=0.1 Btu/hr-ft² curve to demonstrate the effectiveness of a discrete improvement (50 percent) in the state of the art. Based on this method of CRM comparison, all the conceptual reference missions that have been mentioned are within the current state of the thermal-performance art. However, this concept is based on theoretical thermal performance only. The fact that a CRM is feasible when appraised in this manner does not mean that the CRM is feasible when all factors are considered. Such factors as current manufacturing techniques and current materials constrain the apparent capability that results from a consideration of theoretical thermal performance alone. Thus, when all system factors are considered, most of the conceptual reference missions are not feasible when only the metabolic-oxygen requirements are appraised. When power and propulsion requirements are added to the concept, even fewer conceptual reference missions are feasible currently. For example, the Q_L/A value of 0.2 Btu/hr-ft^2 is representative of the Apollo CGSS oxygen tank which contains 323.5 pounds of liquid oxygen within a volume of 4.7 ft^3. Conceptual reference mission IIA, the CRM that involves the smallest amount of cryogen of any of the conceptual reference missions considered herein, involves 2151 pounds of liquid oxygen for metabolic purposes only. The volume that would be required to store that much liquid oxygen is 31.8 ft^3; this is a tank that is approximately 6.7 times as large as the Apollo tank. Currently there are no manufacturing facilities that have the capability to fabricate a dewar this large (that is, an exact scale-up). Nor do the techniques and processes exist that are required for fabrication of such a tank. Even if the fabrication capability existed, size effects alone would be constraining. If the value of Q_L/A could be reduced by an order of magnitude to approximately 0.02 Btu/hr-ft^2, even the most complex conceptual reference missions discussed herein would be within cryotechnological capability.

Summary

Part 3 of this chapter included a discussion of some of the problems that might be encountered in efforts to store cryogenic fluids on or under the lunar surface. The use of these cryogens for power generation and metabolic-gas supply in conjunction with a proposed manned lunar farm and laboratory was discussed. It has been shown that lunar subsurface storage of cryogenic propellants is not as efficient as might be assumed because of the loss of thermal equilibrium in low- and high-boiling-point propellants. In part 2, semidetailed examples of systems-engineering methods applied to the aerospace cryogenic problems of near-future missions were discussed. Although part 3 is less detailed than part 2, it is a preliminary systems-engineering approach to the definition of the metabolic-oxygen requirements for an advanced lunar base and for several advanced planetary missions. These far-future missions were characterized as conceptual reference missions. It was noted that improvement is needed in the state of the cryogenics art before the more complex conceptual reference missions will be feasible from the cryotechnological viewpoint. It was suggested that an order-of-magnitude improvement in the value of heat leak per unit area (Q_L/A) from 0.2 to 0.1 Btu/hr-ft^2 to approximately 0.02 to 0.01 Btu/hr-ft^2 would be such a significant improvement in the state of the art that most of the conceptual reference missions would be feasible.

4 Diversified Applications of Cryogenics Technology

In the 1840's, John Gorrie, a Florida physician, developed an expansion engine for refrigeration in an attempt to alleviate the discomfort of malaria patients. In 1851, Gorrie patented an ice machine which was based on the principle of cooling by expansion (ref. 16). Historically cryogenics has been associated with medicine. Therefore, the medical applications of cryogenic methods that have been developed in the aerospace industry simply are a renewal of a relatively old association. The use of gases in the modern hospital is indispensable and most medical gases are available in cryogenic form.

HOSPITAL APPLICATIONS

Generally the hospital use of gases that are available as cryogens can be divided into three categories based on use: bedside use, surgical and radiological use, and laboratory use. The bedside use of oxygen tents or masks is a familiar application of bottled oxygen. Surgical gases include oxygen, nitrogen, and carbon dioxide, all of which are available in cryogenic form. These cryogens may be used in conjunction with inhalant anesthetics or in cryosurgical probes. Also, oxygen may be used as an inhalant to increase tissue radiosensitivity prior to radiotherapy. In addition to oxygen, nitrogen, and carbon dioxide, such gases as hydrogen and argon are used in the biomedical laboratory. Applications include gas chromatography, histologic techniques, and other uses that are too numerous to mention. The purity, cost, safety, and ease of handling a cryogen compared with handling the same gas in high-pressure storage bottles makes feasible the use of cryogens even in the small hospital or clinic.

Large hospitals have a metabolic-oxygen use rate that is sufficiently great to make feasible the installation of a large-volume facility-type liquid-oxygen (LO_2) dewar. These dewars may be buried conveniently, and because no weight penalty is involved, as is the case for flight systems, insulation is not problematic. In this configuration, the gas must be delivered through a hospital-wide distribution system. Usually such a system is incorporated in the original design and construction. The hospital of smaller size, which has a lower metabolic-oxygen use rate, either must install the system just described (at significant expense and inconvenience) or must use the traditional high-pressure storage bottles. For a hospital in this situation, another alternative exists: a subcritical cryogenic gas storage system (CGSS) in the form of a depot dewar. The following is a description of the general configuration of such a system and an economic-feasibility estimate.

At a site remote from the bedside, surgical and radiological suites, or the laboratory, a depot can be maintained for storage of the gases that are used most frequently (fig. 81). The depot could be a colony of subcritical dewars. The dewar on which these calculations are based is the 41.5-in.-o.d. 56-day Apollo Applications Program (AAP) dewar (also called the Skylab dewar). The capacity of the dewar is approximately 1200 pounds of liquid oxygen. If left undisturbed at 70° F, the system would lose fluid by boiloff at the rate of 5.2 pounds per day. However, if the depot environmental temperature is reduced to 35° F by means of mechanical refrigeration, the boiloff loss rate is reduced to 2.8

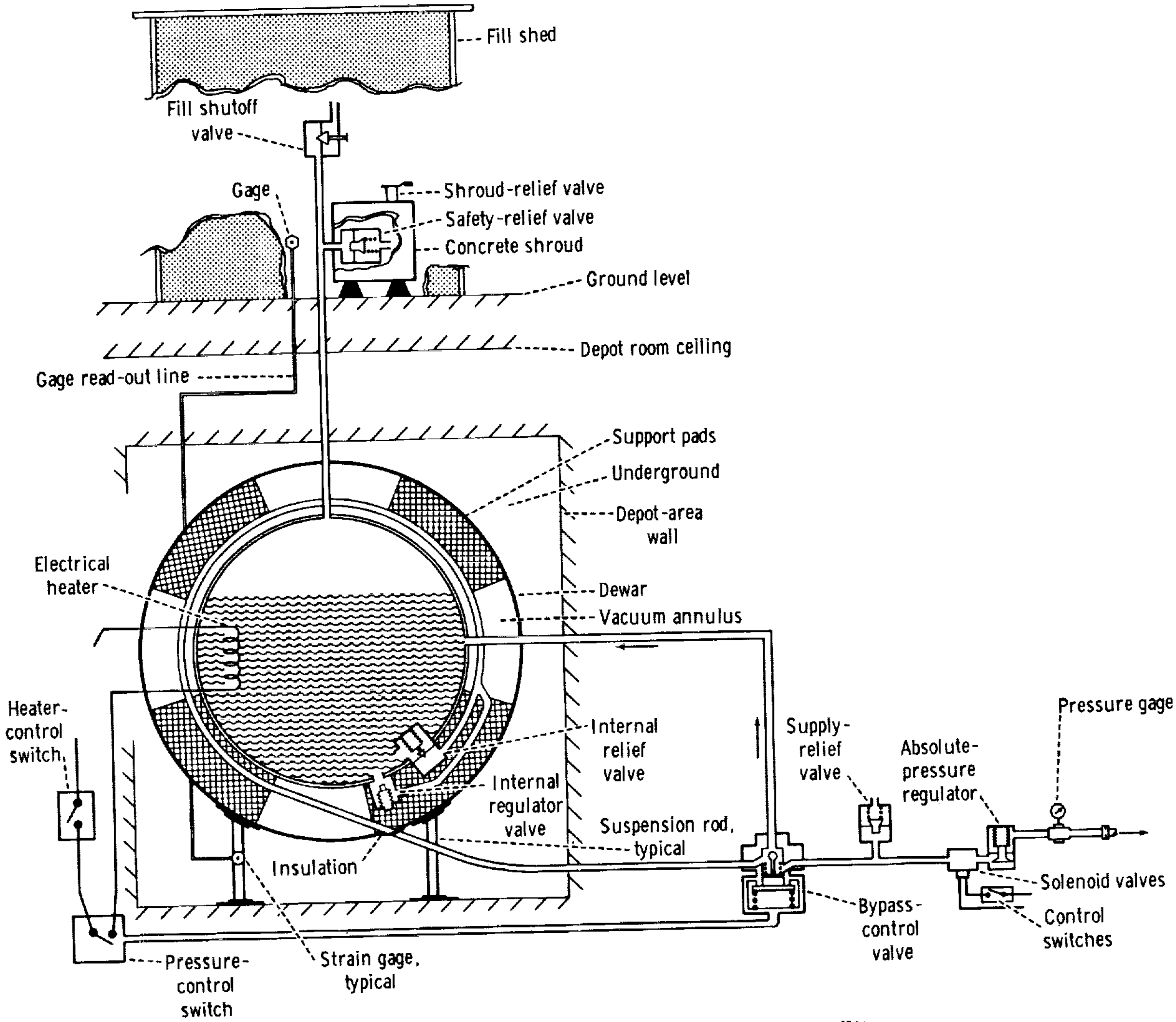


FIGURE 81.—Typical liquid-oxygen dewar depot facility.

pounds per day. The quantity of liquid oxygen that remains in 41.5-in.-o.d. depot dewars as a function of time is shown in figure 82. Also, unrefrigerated and refrigerated depots are compared in figure 82. This boiloff-rate reduction is predicated on conditions whereby the vented gas is lost into the atmosphere; but in a refrigerated depot, the quantity that is boiled off would be less than the value calculated in the preceding example. Obviously the bigger the hospital considered, that is, the larger the number of patients (N_P) that are using oxygen, the larger the number of dewars that are needed to meet the demand. The number of patients using oxygen is plotted in figure 83 as a function of resupply interval (fill lifetime) for both one and two 41.5-in.-o.d. depot dewars. In figure 83, the center region of each curve is not necessarily representative of a path function but represents the regions of the curves that are constrained thermally. Above and below these regions, the curves are constrained by the time of use and the number of patients that are using the gas. Waste by means of boiloff is negligible. For zero time, an infinite number of patients could be served; for zero patients, the equilibrium flow rate will prevail. For any number of patients ($N_P>0$), demand-flow conditions will prevail.

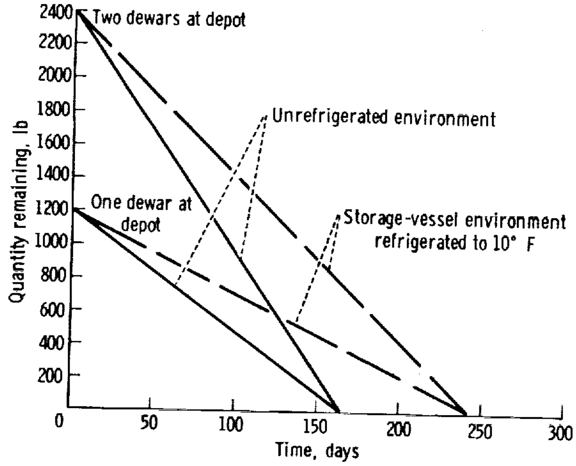


FIGURE 82.—Quantity of oxygen remaining in 41.5-in.-o.d. depot dewars as a function of time (loss is because of vented boiloff only).

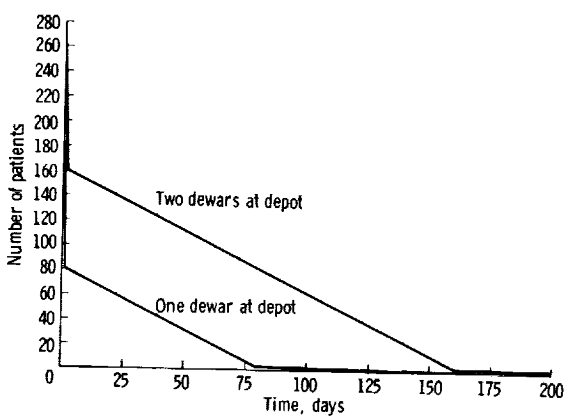


FIGURE 83.—Number of patients supplied with oxygen as a function of resupply interval for the central depot (41.5-in.-o.d. dewars).

Piping the gas to the use site will not be considered for a small or medium-size hospital. In such a hospital, a 20- to 30-in.-o.d. dewar could be maintained as a ward dewar or regional dewar. This dewar could be connected directly to the use site (oxygen tent or mask, anesthesia-control station, and so forth). When empty, the ward dewar can be wheeled to the depot, connected to the depot-dewar effluent line, and filled with liquid oxygen. Then the ward dewar can be disconnected, wheeled to the ward-dewar room, and either put online or placed in standby configuration. The dewar-handling procedures are safer than high-pressure-bottle handling procedures, and, because of the longer resupply interval that is involved in the dewar system, the procedure is performed less frequently. A generalized concept of how such a dewar could be used in an intensive care unit is shown in figure 84 and a detailed view of the oxygen-line route is shown in figure 85. A 41.5-in.-o.d. and a 23-in.-o.d. dewar are used because of the availability of data on these tanks. It is not to be implied that these two tank sizes are suitable for any particular hospital installation. If a smaller dewar (for example, a 12-in.-o.d. vessel) is considered, the results are as shown in figure 86. This figure is indicative that a 12-in.-o.d. clinical dewar would not be economical; that is, a larger dewar should be used. These use-rate calculations were based on an oxygen use rate of 8.0 liters per minute per patient at 1 atmosphere and at room temperature. This use rate is equivalent to 33.12 pounds per 24 hours at a pressure of 1 atmosphere. The cost to one hospital is $4.50 per 244 cubic feet of oxygen. This volume is equivalent to 17 pounds of liquid oxygen or one high-pressure storage bottle. At a use rate of 8.0 liters per minute per patient, two high-pressure bottles of oxygen are needed every 24 hours, at a cost of $9.00 per patient per day. At the same use rate per patient, liquid oxygen would cost $0.51 per patient per day (based on a cost rate of $0.03 per pound of liquid oxygen). The equivalent mass-storage capacities of high-pressure storage bottles and cryogenic dewars are compared in figure 87. As shown in figure 88, it is more economical to use cryogenic oxygen than bottled oxygen. The subcritical cryogenic gas storage systems involve relatively infrequent servicing and are much safer and easier to manipulate than are high-pressure storage bottles.

CRYOBIOLOGY AND CRYOSURGERY

The two cryobiological topics that have received extensive attention recently are cryosurgery and the application of cryogens to low-temperature phenomena such as the freeze drying of microbial cultures, lyophilization in the preparation of vaccines, and the

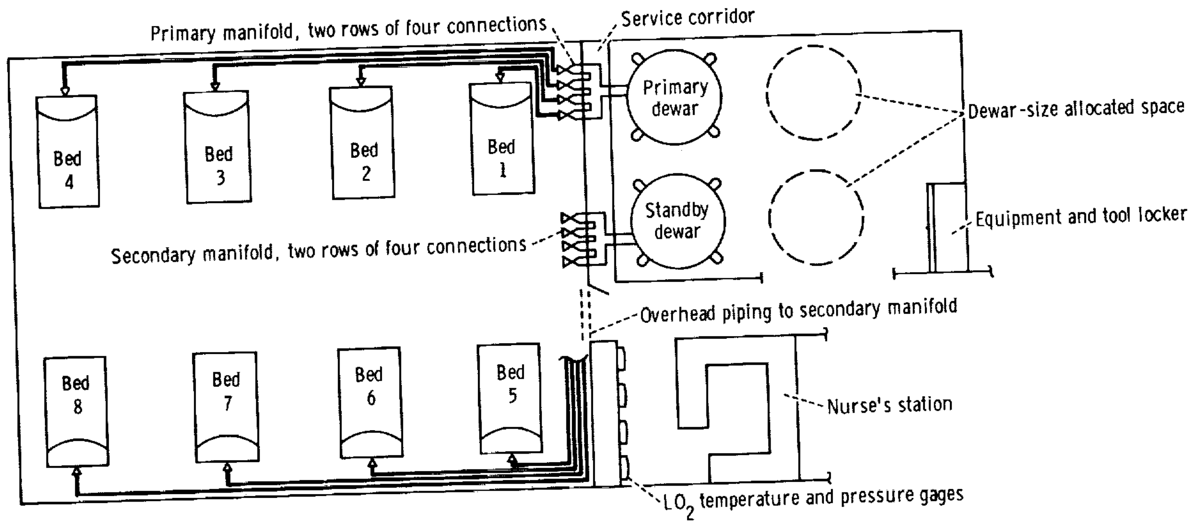


FIGURE 84.—General view of the regional oxygen-delivery system.

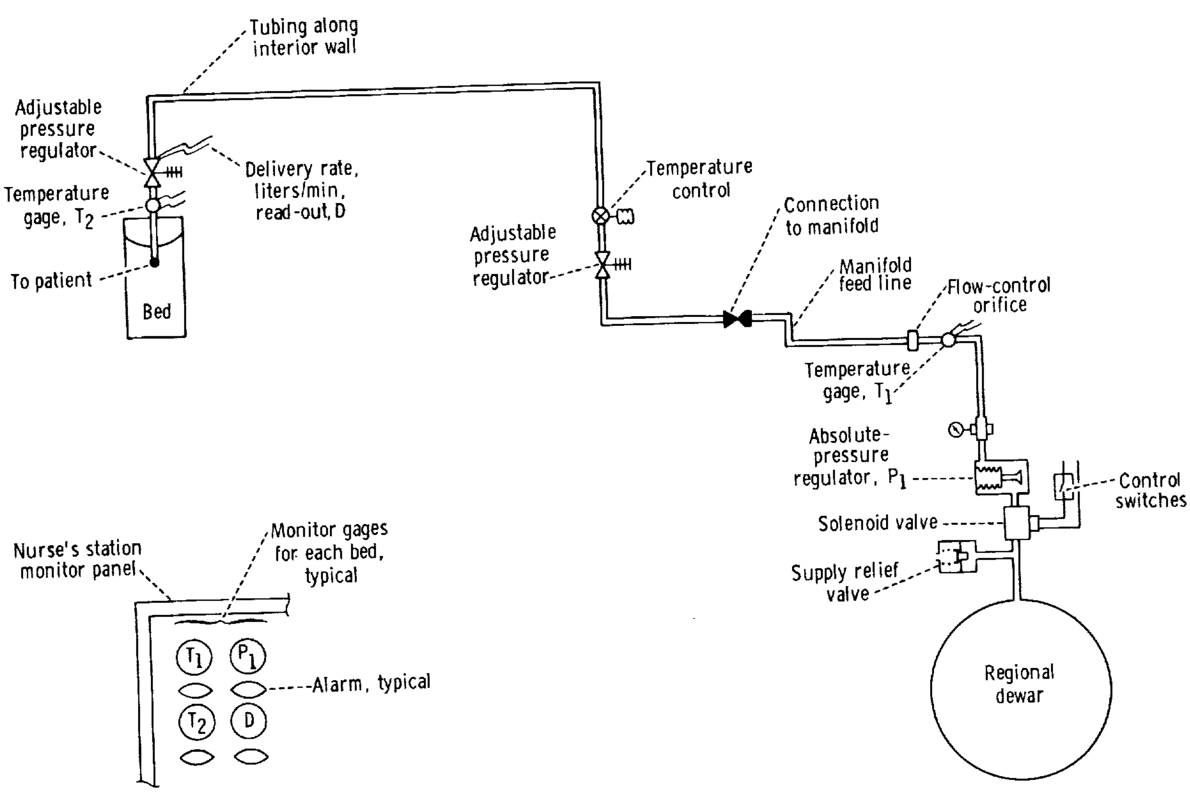


FIGURE 85.—Detail of the regional oxygen-delivery system.

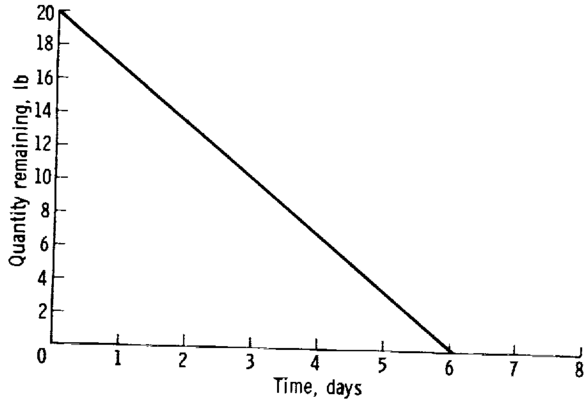


FIGURE 86.—Fill lifetime of one 12-in.-o.d. clinical dewar; loss is caused by vented boiloff only.

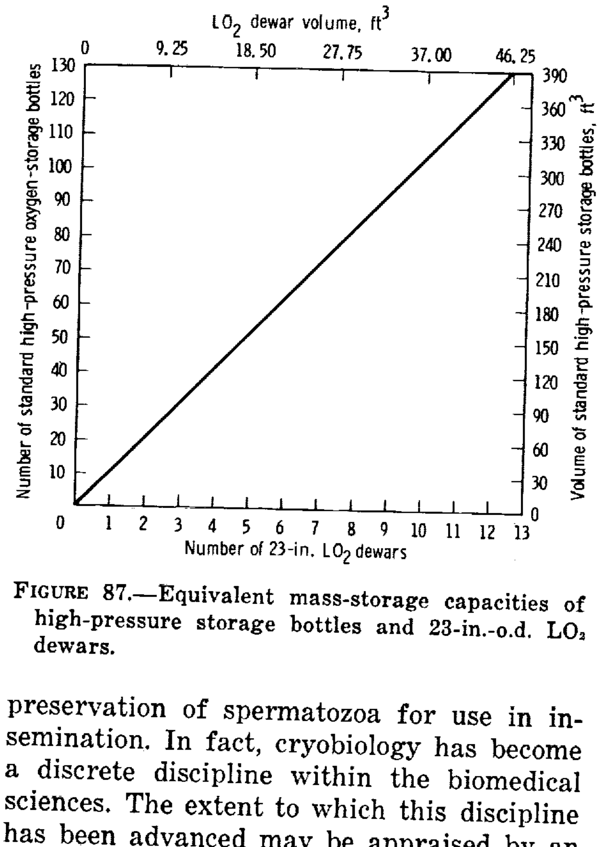


FIGURE 87.—Equivalent mass-storage capacities of high-pressure storage bottles and 23-in.-o.d. LO_2 dewars.

preservation of spermatozoa for use in insemination. In fact, cryobiology has become a discrete discipline within the biomedical sciences. The extent to which this discipline has been advanced may be appraised by an inspection of a typical example of the literature (ref. 17). Cryobiology is too extensive for discussion here; however, a few representative topics will be discussed briefly in this section.

Several thermal regions are involved in the

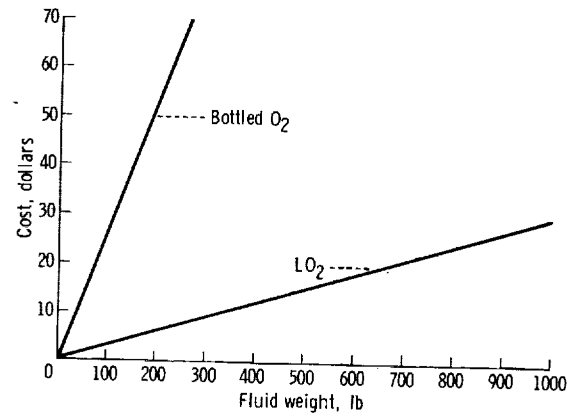


FIGURE 88.—An economic comparison of bottled oxygen and cryogenic oxygen for hospital use.

process of freezing a biological system. A curve that is descriptive of this process is shown in figure 89, which has been reproduced from reference 18 by permission of the copyright owner (McGraw-Hill Book Company). First, sensible heat is removed from the liquid phase that is inside the material. Second, latent heat is removed and the liquid phase is frozen. Third, after solidification, sensible heat is withdrawn from the solid phase until steady state is achieved. There is a critical zone of temperature, between 32° and −70° F, through which the specimen must be cooled rapidly or the biological system will be damaged by ice crystals. Whenever possible, biological specimens should be small so that radial temperature gradients may be minimized.

The use of cryogenics technology in medicine is becoming more and more extensive. Cryosurgical procedures are used most extensively in dermatology and neurosurgery. However, cryosurgical methods are used in ophthalmology for correction of retinal detachment (cryopexy) and in otology for certain inner-ear procedures. Most dermatologic cryosurgical procedures are relatively simple. Surgery of the integument does not involve extensive preliminary work to make the site of interest accessible. Thus the use of cryogens such as liquid nitrogen (LN_2) for the removal of many types of cutaneous lesions is uncomplicated from the surgical viewpoint.

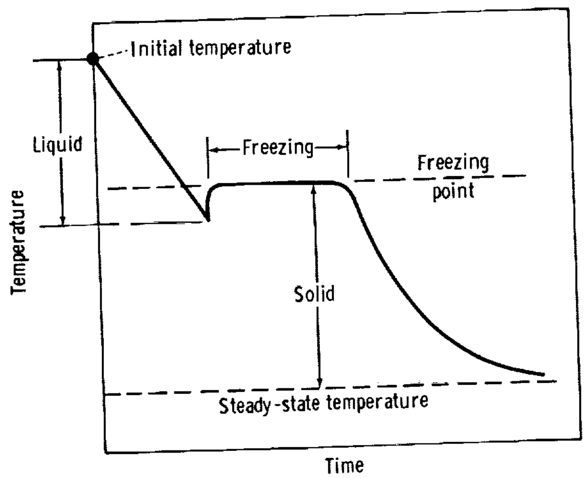


FIGURE 89.—Freezing curve for a biological system. From Cryogenic Systems by Randall Barron, 1966. Used by permission of McGraw-Hill Book Company.

Usually these methods produce good cosmetic results. Also, clinical dewars are easy to fill, maintain, and use; and the cost is nominal. The Cooper cryosurgical system, reproduced from reference 18 by permission of the copyright owner (McGraw-Hill Book Company), is shown in figure 90. A thermocouple attached to the probe tip is used to monitor the tip temperature. A general-purpose cryosurgical system, also reproduced from reference 18 by permission of the copyright owner (McGraw-Hill Book Company), is shown in figure 91. A thermocouple is used to monitor tip temperature, and a miniature Joule-Thomson device is used for refrigeration.

Neurosurgical procedures that involve a cryogen such as liquid nitrogen are much more complex than are the dermatologic procedures just described, but are much simpler than some of the older neurosurgical methods. The exposure of the intracranial site of interest often is complicated and is not without danger to the surrounding (normal) tissue. Such complications and dangers are minimized by the use of cryosurgical methods. Of course, cryogenic methods are not applicable in all procedures. Many operational advantages are associated with cryogenic methods. Hemorrhage is minimized and the methods are applicable rapidly, minimizing risk to the

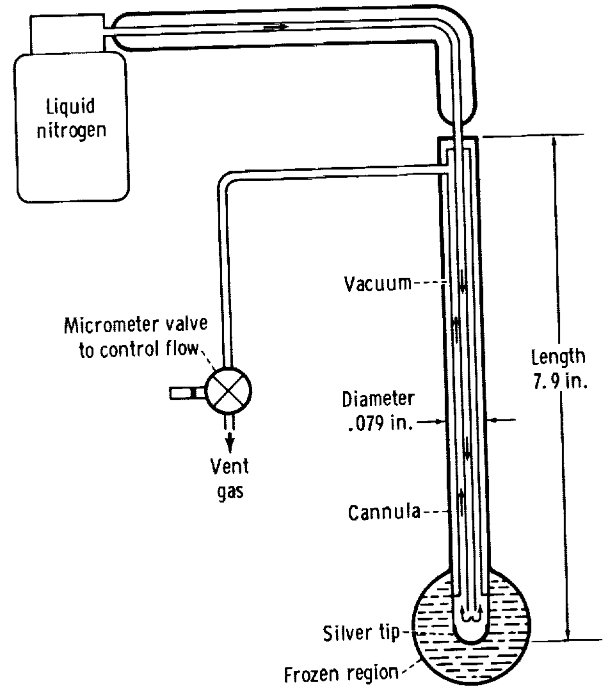


FIGURE 90.—Cooper cryosurgical system. From Cryogenic Systems by Randall Barron, 1966. Used by permission of McGraw-Hill Book Company.

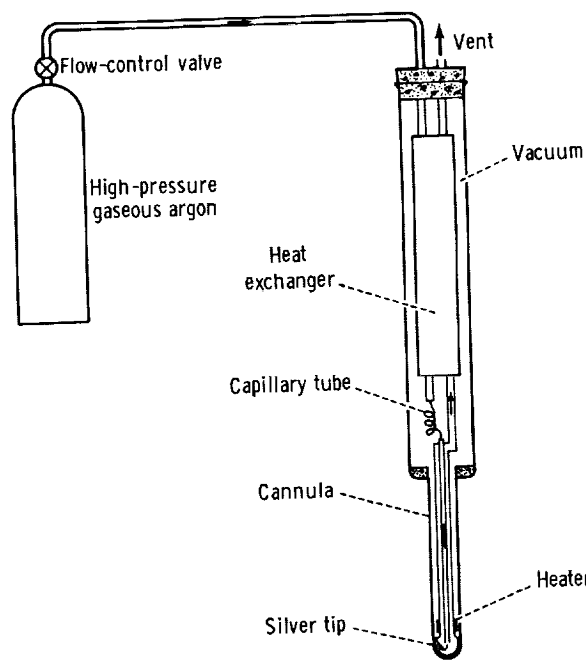


FIGURE 91.—General-purpose cryosurgical system. From Cryogenic Systems by Randall Barron, 1966. Used by permission of McGraw-Hill Book Company.

patient. The procedures are reproducible consistently, can be delimited accurately, and are reversible, enabling the surgeon to observe the consequences of a particular lesion before the tissue is destroyed (a feature that is of great importance in neurosurgery). Also, cryogenic methods are flexible: The quantity of tissue destroyed can be varied and controlled, another particularly significant factor in neurosurgery.

Cryothalamectomy for Parkinsonism is a good example of all the advantages of cryosurgical methods put to use in a single procedure. The probe tip is cooled to approximately 14° F and is allowed to remain in a given location in the thalmus for as long as 30 seconds. By observation of the motor responses of the patient, the neurosurgeon can determine when the probe is located properly. Cerebral tissue cooled to 14° F can be thawed with no adverse consequences if thawing occurs less than 30 seconds after chilling. When the pathologic focus has been located precisely, the tip is cooled to −40° to −60° F for approximately 3 minutes. After destruction of the pathologic focus, the tip is warmed rapidly and is withdrawn. A sphere of dead tissue that is approximately 0.25 inch in diameter is the result of cryothalamectomy. The sphere of tissue becomes encysted and is innocuous to the patient. One study (ref. 19) that involved this procedure resulted in a 93-percent cure rate in 3000 patients operated upon, and there were no side effects. A review of cryosurgery is given in reference 20.

DEEP SUBMERGENCE SEARCH VEHICLE

The Deep Submergence Search Vehicle (DSSV) is being developed by the U.S. Navy for use in conjunction with a nuclear (submarine) mother ship (NMS). It has been noted (ref. 21) that 97 percent of the ocean floor can be reached by the use of such a submersible vehicle, and that mission times may exceed 30 hours, which is approximately a threefold endurance increase over the capability of current submersibles. The uses of such a vehicle in oceanography, marine biology, undersea communications-equipment installation and maintenance are numerous and are obvious.

A fuel-cell system (FCS) will be the primary power generation source and batteries will be used as an auxiliary power source. The hydrogen/oxygen fuel cell that was developed for aerospace applications produces electrical power, water, and heat. It has been recognized (ref. 21) that the hydrogen/oxygen fuel cell in the DSSV would be operating in sea water (a vast source of potential reactants) and that a support module can be used to purify and decompose sea water. Donaldson (ref. 21) has compared the DSSV and the advanced aerospace mission requirements for the reactants supply subsystem

TABLE 37.—*Deep Submergence Search Vehicle Reactant Supply Subsystem Oxygen Assembly Characteristics* [a]

[DSSV compared with advanced aerospace mission requirements]

Constraints	Aerospace	DSSV
Mating and replenishment		
Ambient conditions	Ambient air	Sea water[b]
Replenishment time	Optional	Short[b]
Vent during fill	To ambient air	To storage[b]
Operational		
Storage quantity, lb	600	543
Nonvented standby, hr	[b] >48	<4
Maximum delivery rate, lb/hr	[b]18	40
Minimum delivery rate, lb/hr	[b]0.56	5
Purity of reactants delivered	High	Very high[b]
Environmental		
Ambient pressure		
Fill mode, psia	20	[b]20
Operating mode, psia	0	[b]8900
Dynamic loading		
Gravity levels	Very high[b]	Low
Duration, min	[b] <15	$>10^5$

[a] Reprinted from Undersea Technology, December 1968, page 33, by permission of copyright owners, Compass Publications, Inc. (ref. 21).
[b] Design-limiting factors.

(RSS) oxygen assembly; these data are given in table 37. Reactants will be stored at cryogenic temperatures. A notable weight advantage results from the uses of a cryogenic storage system instead of high-pressure storage bottles. It has been observed (ref. 21) that the aerospace reactants supply subsystem is a complex storage system with controls, whereas the hydrospace reactants supply subsystem is a complex fluid-transfer system with associated storage vessels.

A typical power installation for the DSSV is shown in figure 92, and a model of the DSSV attached to the NMS is shown in figure 93. These figures have been reproduced from reference 21 by permission of the copyright owners (Compass Publications, Inc.).

A detailed analysis of the operational aspects of the cryogenic storage system and hydrogen/oxygen fuel cell is beyond the scope of this report. A generalized concept of the DSSV/NMS interface is shown in figure 94. However, a brief summary of the operation of the systems as defined by Donaldson (ref. 21) is given in the following paragraph.

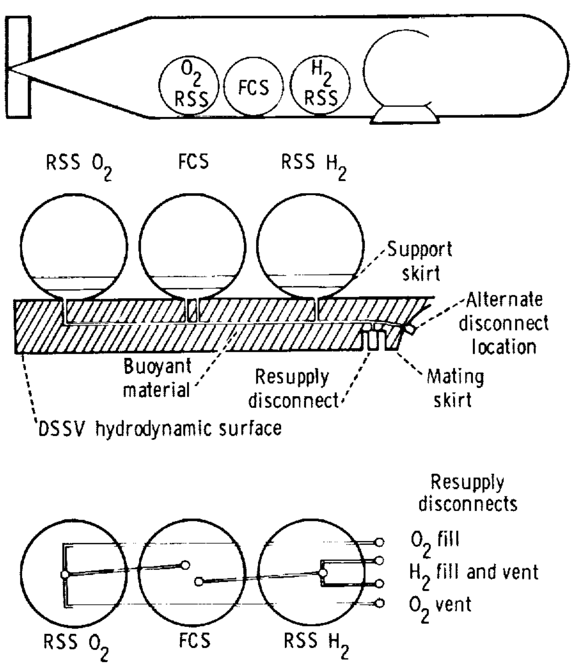


FIGURE 92.—The DSSV power installations. Reprinted from Undersea Technology, December 1968, page 32, by permission of copyright owners Compass Publications, Inc. (ref. 21).

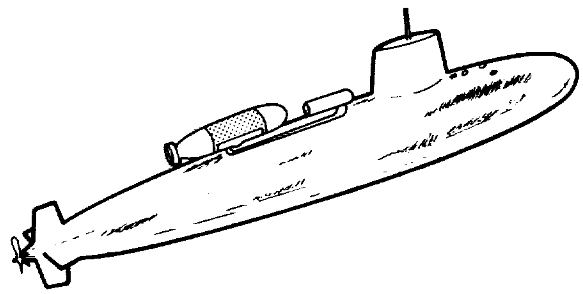


FIGURE 93.—Model of the DSSV mounted aft on a model of the nuclear submarine mother ship. Reprinted from Undersea Technology, December 1968, page 32, by permission of copyright owners Compass Publications Inc. (ref. 21).

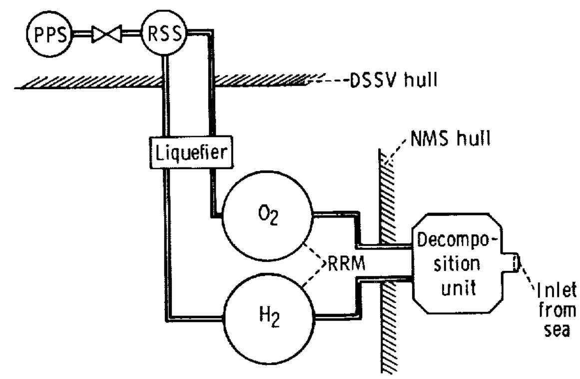


FIGURE 94.—Generalized concept of DSSV/NMS interface.

The power production subsystem (PPS) and the reactants supply subsystem are interconnected and are part of the DSSV. The reactant supply module is located on the aft deck of the support vehicle. When resupply is in progress, the flow of reactants is from the reactants resupply module (RRM) to the reactants supply subsystem, and the power production subsystem is isolated by closure of the valve in the line from the reactants supply subsystem to the power production subsystem. When the DSSV is independent of the NMS, the reactant flow is from the reactants supply subsystem to the power production subsystem. The liquefier is part of the NMS equipment. Decomposition products are stored in the reactants resupply module, and the decomposition unit can be opened directly to the sea.

SUMMARY

From the point of view of technology spinoff, there are several significant factors that must be discussed. Operational considerations include such factors as system reliability, system efficiency, safety, and so forth. Economic considerations include such factors as development cost, initial cost, service cost, and so forth. One of the advantages of the technology spinoff in cryogenics is that the development costs for current state-of-the-art hardware and technology have been paid; only other costs remain to be defrayed. Of course, manufacturing costs will include some costs of a developmental nature (for example, minor design modification to achieve manufacturing economy or operational optimization).

During the developmental phase, operational considerations are chosen and economic considerations are deduced therefrom. In the adaptation of aerospace-developed systems to nonaerospace uses, the economic parameter becomes of paramount importance. The increased efficiency or performance capability that the new system involves must be justifiable in terms of its implementation cost. Cryogenic gas storage systems are an excellent example of aerospace systems that can be derated to such a level that they become both operationally and economically suitable for nonaerospace applications. For example, great development effort was expended to achieve very thin tank walls for the purpose of weight reduction. These thin walls are a hazard in nonaerospace use because they are subject to denting and other forms of damage. However, the tank walls can be made thicker for nonaerospace use, making fabrication more economical and improving an aerospace system characteristic that is undesirable in systems that are destined for nonaerospace use.

The significance of technology spinoff from aerospace cryogenics has been illustrated by means of several examples of diversified nonaerospace applications. These applications included a hospital system for metabolic-oxygen supply to patients and for the supply of other cryogenic gases for other biomedical applications. The feasibility of such an installation was discussed in economic, medical, safety, and engineering terms. Also, the use of CGSS in a submersible vehicle was discussed as another nonaerospace application of aerospace cryogenics technology. Manned submarine vehicles involve problems of metabolic-gas storage and cryogenic power-generation-gas storage that are similar to problems that involve the corresponding systems on manned spacecraft. A design concept was used to illustrate this point. The nonaerospace use of aerospace cryogenic gas storage systems was put into perspective with respect to operational characteristics and development, fabrication, and service costs. From the preceding factors, it may be concluded that cryogenics technology spinoff from the space program has resulted in advancements in several diversified nonaerospace fields.

Concluding Remarks

In the last century, cryogenics has progressed from a laboratory experiment to a practical and useful servant of man. The majority of that progress has been achieved in the last few years. The key to this rapid progress and to the successful production of cryogenic gas storage systems that can operate unattended in the isolated and hostile environment of space was found in the systems-management approach. In retrospect, the development and verification of the systems-management approach to achieving success in large-scale programs may be one of the most important results of the space program.

The newly acquired ability of man to cooperate successfully in complex programs has been discussed elsewhere. However, the application of this approach is the concept that unifies each of the topics that are described in this report. In chapter 1, the early milestones of cryogenics are summarized in chronological order. Chapter 2 contains a discussion of recent and current cryogenic gas storage systems for use in manned space flight. In chapter 3, future missions are discussed from the point of view of cryogenics requirements: the near-future-missions section includes some mission concepts that now are under consideration; the far-future-missions section includes example missions that may be only in the feasibility-study phase at the present time. In chapter 4, some arbitrarily selected examples of the use of cryogenic gas storage systems in nonaerospace technology are discussed.

A broad technological base has been established that will be useful to mankind and for future space programs. The knowledge that has resulted from technology-development programs has only begun to affect the daily life of man. The knowledge that has been gained in science and systems management will be applied to make future space programs possible and to improve the quality of life on Earth. Perhaps the challenges that lie ahead are greater than any others that have been confronted by man.

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Appendix—Fluid Thermodynamics

After filling and capping a cryogenic gas storage system, a standby period of several hours may be required. During standby, the stored cryogen is receiving heat energy, which pressurizes the stored fluid. The system can be either nonvented or vented during standby. Nonvented standby time is the time from capping a vessel until venting from the relief valve occurs. During nonvented standby, it is assumed that no venting will occur prior to use of the fluid. Vented standby is based on the assumption that the system will vent at some predetermined pressure prior to use of the fluid. The selection of vented or a nonvented system depends on the mission, standby time, weight, effect of vapor cooling, ambient temperature, vessel volume, and system design.

THERMODYNAMIC EQUATIONS

For a vented system, the heat that is required to pressurize the vessel is given in equation (A-1)

$$Q = WH_A + M_f H_f - M_i H_i - M_f P_f V_f + M_i P_i V_i \qquad \text{(A-1)}$$

where Q = heat energy, Btu/lb
M = stored mass, lb
H = enthalpy, Btu/lb
P = pressure, lb/ft²
V = specific volume, ft³/lb
W = flow rate, lb/hr
A = average of initial and final conditions
f = final fluid condition
i = initial fluid condition

For a nonvented system, the flow term WH_A in equation (A-1) is zero, because the system is only building up pressure. Therefore, for a nonvented system, the heat that is required to pressurize the vessel is given in equation (A-2).

$$Q = M_f H_f - M_i H_i - M_f P_f V_f + M_i P_i V_i \qquad \text{(A-2)}$$

Initially, after filling and capping, the fluid is in two distinct phases (liquid and gas), and the total stored mass can be expressed as shown in equation (A-3).

$$M_f = M_l + M_g \qquad \text{(A-3)}$$

where g = gas
l = liquid

If the volume fraction of the liquid or gas in the system is F, then equations (A-4) and (A-5) can be substituted for the initial conditions.

$$F_l = \frac{M_l V_l}{M_f V_f} \qquad \text{(A-4)}$$

$$F_g = \frac{M_g V_g}{M_f V_f} \qquad \text{(A-5)}$$

Substitution of equations (A-3), (A-4), and (A-5) into equation (A-2) results in equation (A-6).

$$Q = M_f H_f - \left(\frac{F_l M_f V_f}{V_l} H_l + \frac{F_g M_f V_f}{V_g} H_g\right) - M_f P_f V_f + \left(\frac{F_l M_f V_f}{V_l} P_l V_l + \frac{F_g M_f V_f}{V_g} P_g V_g\right) \qquad \text{(A-6)}$$

The heat input that is required to pressurize a supercritical vessel as a function of pressure and percent fill for oxygen and parahydrogen is shown in figures A–1 and A–2. After a storage system has been pressurized, the pressure must be maintained during operational fluid delivery. The heat required to maintain the operating pressure in a subcritical system can be determined from the energy-balance equation. The energy that is required to maintain pressure in a supercritical system is expressed by equation (A–7).

$$Q=-D\left(\frac{dH}{dD}\right)_p \qquad \text{(A–7)}$$

where D = density, lb/ft^3
p = constant pressure

The solution of equation (A–7) yields the heat energy that is required to maintain pressure per pound of fluid withdrawn as a function of fluid density.

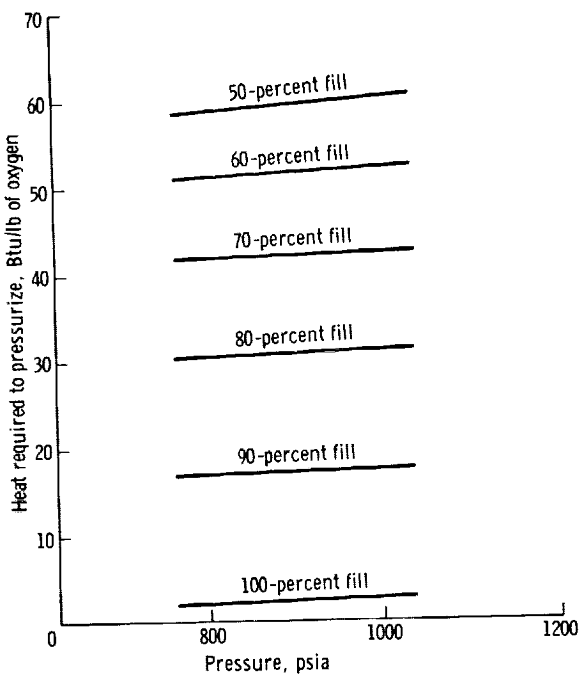

FIGURE A–1.—Plot of the heat required to pressurize a supercritical vessel as a function of pressure at various percentage fills with oxygen.

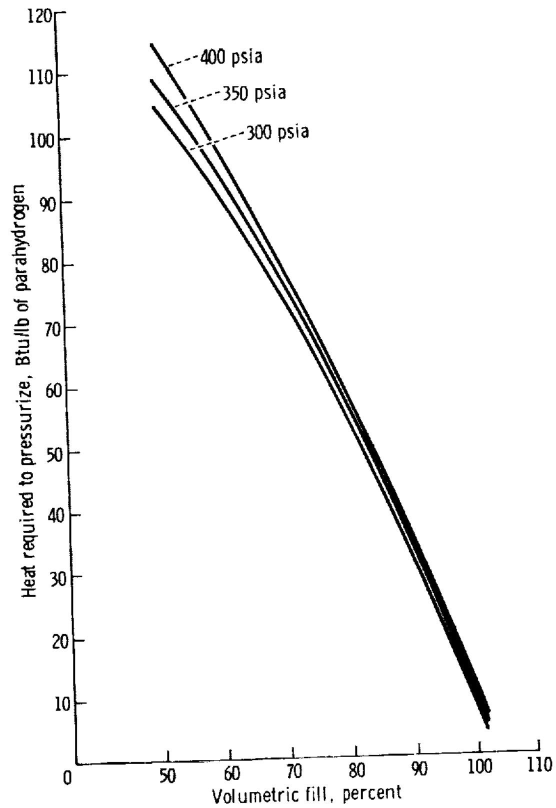

FIGURE A–2.—Plot of the heat required to pressurize a supercritical vessel as a function of percentage fill with parahydrogen.

The total heat input is comprised of heat from the internal heater, environmental heat input, electrical sensors, leads, motors, and any other heat sources that are associated with the storage system. Then the required heat energy must maintain the fluid in equilibrium at the desired pressure during operation. The heat that is required for maintenance of constant pressure during operation for various storage pressures is shown as a function of fluid density for oxygen and parahydrogen in figures A–3 and A–4.

Because specific missions require certain cryogen flow rates, it is desirable to design the vessel heat-input rates for the required mission flow rates. That is, a vessel should be designed thermally so that its expelled fluid is utilized and is not wasted by venting because of excessive pressurization. Various fluid-expulsion rates are shown as a function

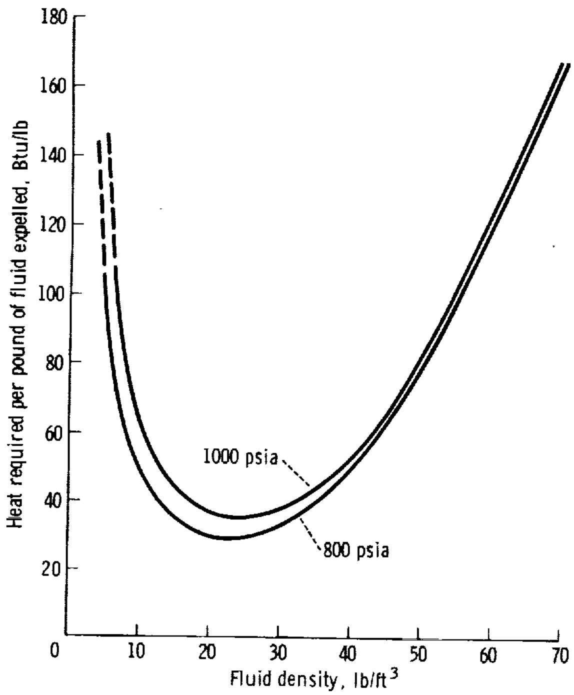

FIGURE A-3.—Plot of the heat required to maintain pressure per pound of oxygen (supercritical) withdrawn.

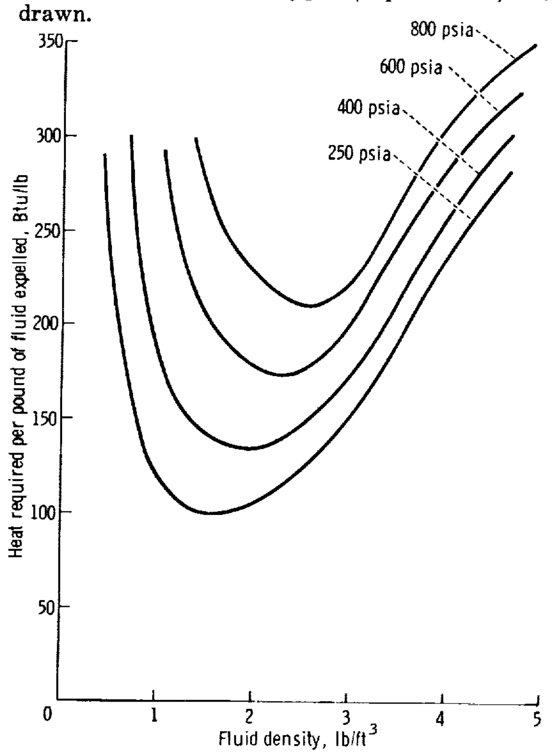

FIGURE A-4.—Plot of the heat required to maintain pressure per pound of hydrogen (supercritical parahydrogen) withdrawn.

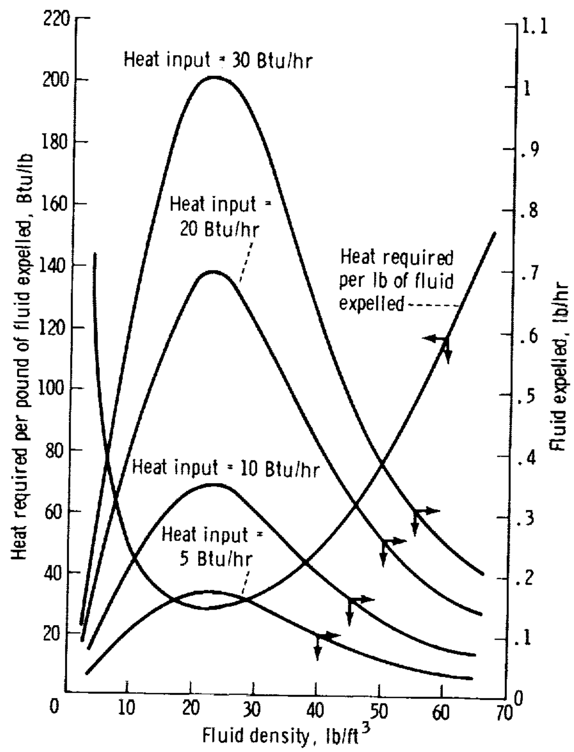

FIGURE A-5.—Plot of the heat required to maintain pressure per pound of oxygen (supercritical, 800 psia) withdrawn and the fluid quantity expelled at four heat-input rates.

of heat input and fluid density for oxygen and parahydrogen storage systems in figures A–5 to A–8. Cryogenic properties of several gases are shown in table A–1.

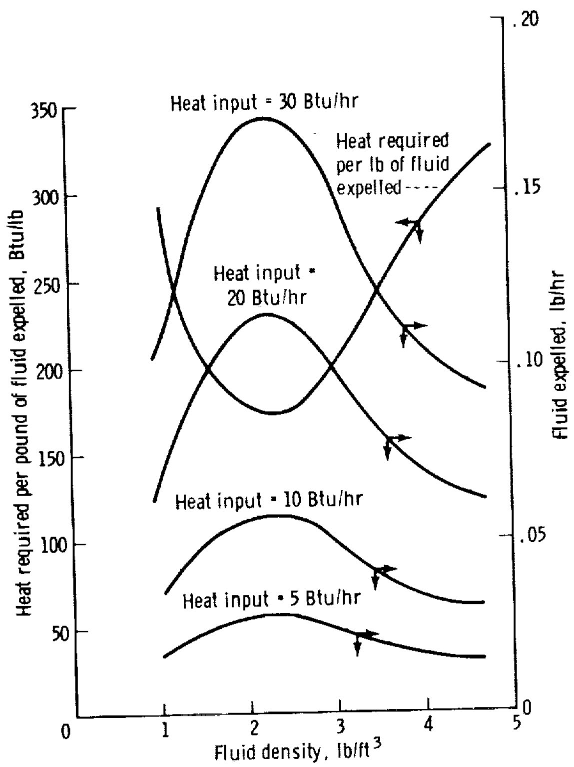

FIGURE A–6.—Plot of the heat required to maintain pressure per pound of hydrogen (supercritical, 600 psia) withdrawn and the fluid quantity expelled at four heat-input rates.

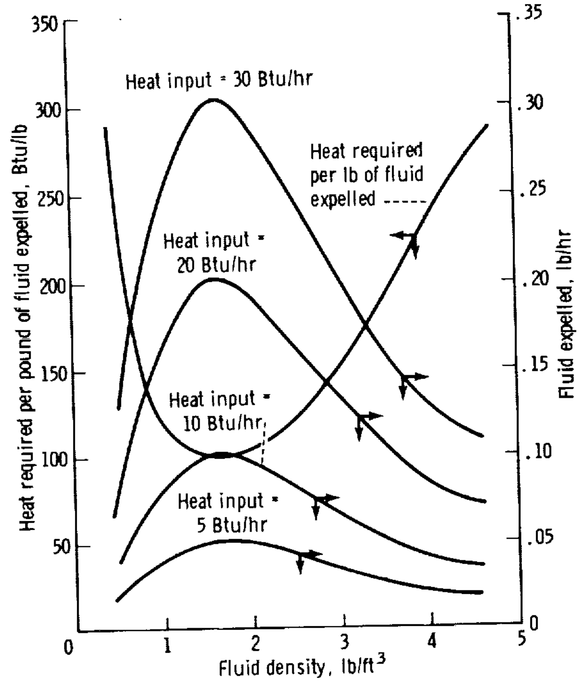

FIGURE A–7.—Plot of the heat required to maintain pressure per pound of hydrogen (supercritical parahydrogen, 250 psia) withdrawn and the fluid quantity expelled at four heat-input rates.

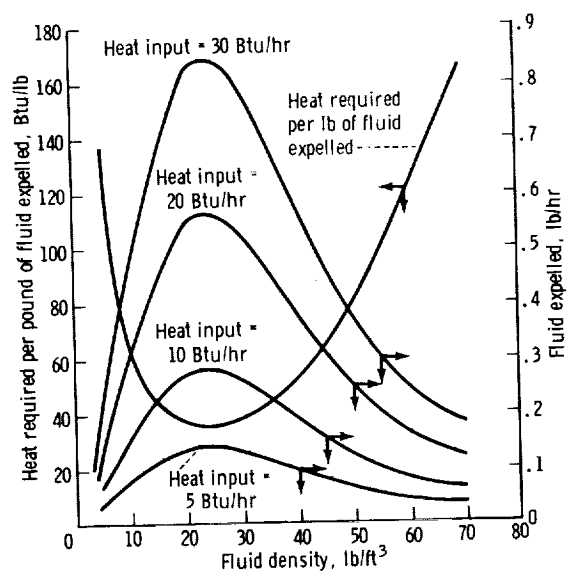

FIGURE A–8.—Plot of the heat required to maintain pressure per pound of oxygen (supercritical, 1000 psia) withdrawn and the fluid quantity expelled at four heat-input rates.

TABLE A–1.—*Cryogenic Properties of Gases*

Property	Gas					
	He	Ne	H_2	N_2	O_2	F_2
Density, 32° F, 1 atm, lb/ft^3	0.01114	0.0562	0.00561	0.0781	0.0892	0.107
Boiling point, 1 atm, °F	−452.0	−410.6	−423.0	−320.4	−297.4	−306.4
Melting point, 1 atm, °F	−458.0	−415.7	−434.4	−345.8	−361.1	−363.0
Vapor density at boiling point, lb/ft^3	1.06	0.593	0.0803	0.2756	0.296	0.352
Liquid density at boiling point, lb/ft^3	7.803	74.91	4.43	50.44	71.17	94.5
Vapor pressure of solid at melting point, mm Hg	<0.02	323	54	96.5	2.0	0.12
Heat of vaporization at boiling point, Btu/lb	8.82	37.4	192.7	85.7	91.6	73.8
Heat of fusion at melting point, Btu/lb	1.8	7.2	25.0	11.0	5.9	5.8
[a] C_p, 59° F and 1 atm, Btu/lb-°F	1.25 (at −292° F)	~0.25	3.39	0.248	0.218	0.180
[a] C_p/C_v, 59° to 68° F and 1 atm	1.66 (at −292° F)	1.64	1.41	1.40	1.40	1.36 (at 70° F)
Critical temperature, °F	−450.2	−379.7	−399.8	−232.5	−182.0	−200.0
Critical pressure, psia	33.2	394.6	190.8	492.3	736.5	808.5

[a] C=specific heat, Btu/lb-°F; v=constant volume; p=constant pressure.

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